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IBM Huntsville

Astrionic System Optimization and Modular Astrionics for NASA Missions After 1974

Preliminary Definition of Astrionic System for Space Tug Mission Vehicle Payload (MVP)

Progress Report

16 June 1970 to 15 August 1970

MSFC-DRL-008

Line Item No. 268

IBM No. 69-K44-0006H

Contract No. NAS 8-14000

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Line Item No. 268

IBM No. 69-K44-0006H

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Astrionic System Optimization and Modular Astrionics for NASA Missions After 1974

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Classification and Content Approval ______________________

Data Manager Approval ______________________

Program Office Approval ______________________

IBM Federal Systems Division, Electronics Systems Center/Huntsville, Alabama

ABSTRACT

The IBM Corporation has been pursuing Instrument Unit Derivative studies under Contract NAS8-14000, Mod. 1586 revised, titled "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974." The purpose is to define Instrument Unit Derivatives which can provide cost effective astrionic systems for numerous programs including space tug, conduct system engineering trade studies which will identify modular elements from which future astrionic systems may be configured, and to perform trade studies relating program requirements, with the objective of defining a minimum family of modular elements.

The space tug is a proposed chemically fueled transportation system which is economical, reusable, multipurpose, and long-lived. It is required to interface with the earth and lunar orbiting space stations, the space shuttle, the reusable nuclear shuttle, bases on the lunar surface, men during EVA on the lunar surface, experiment modules, fueling stations, and satellites. It is a modular vehicle, capable of being reconfigured for a variety of missions. A propulsion module, an astrionic module, a cargo module and a crew module are envisioned. Additional kits such as landing legs and manipulator arms are proposed for special mission requirements.

This report defines the results of preliminary studies to define the tug astrionic system, subsystems, and components to meet the requirements for a variety of possible missions. Most of these elements are packaged in the astrionic module as required by groundrules for space tug studies. The emphasis in this study was to demonstrate the modular astrionics approach in the design of the space tug astrionic system. Considerable analysis of mission requirements is documented to establish "design" missions which fix astrionic system requirements. The space tug operational interface with space shuttle, space stations, and the reusable nuclear shuttle requires transfer of data which must be compatible, whether handled by umbilical or by communication media. The study shows the strong generic relationship of the data management system requirements of these common elements, for the integrated space activity in post 1974.

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LIST OF ILLUSTRATIONS

ABBREVIATIONS AND ACRONYMS

A/D	–	Analog to Digital
AM	–	Astrionic Module
BCIOU	–	Bus Control and Input/Output Unit
BITE	–	Built-In Test-Equipment
BOM	–	Basic Operating Memory
CAU	–	Configuration Assignment Unit
CM	–	Crew Module
CPU	–	Central Processing Unit
CRT	–	Cathode Ray Tube
CSM	–	Command/Service Module (Apollo)
D/A	–	Digital to Analog
DRSS	–	Data Relay Satellite System
DSN	–	Deep Space Network
ECC	–	Error Correction Coding
ECI	–	Externally Controlled Input
ECO	–	Externally Controlled Output
EOS	–	Earth Orbiting Shuttle (Space Shuttle)
EOSS	–	Earth Orbiting Space Station
EVA	–	Extravehicular Activity
GHz	–	Gigahertz
H_2	–	Hydrogen
ICSA	–	Inertial Component Sensor Assembly
IGM	–	Iterative Guidance Mode
I/O	–	Input/Output
IMU	–	Inertial Measurement Unit

ABBREVIATIONS AND ACRONYMS (CONTINUED)

INT-21	–	Intermediate - 21 (Saturn S-IC, S-II and IU Stages)
IU	–	Instrument Unit (Apollo)
KHz	–	Kilohertz
LEO	–	Low Earth Orbit
LH_2	–	Liquid Hydrogen
LM	–	Lunar Module (Apollo)
LOS	–	Line-of-Sight
LOSS	–	Lunar Orbit Space Station
LOX	–	Liquid Oxygen
LR	–	Landing Radar
LRU	–	Lowest Replaceable Unit
MED	–	Multipurpose Electronic Display
MHz	–	Megahertz
MOS	–	Metal Oxide Silicon
MSFN	–	Manned Space Flight Network
MTBF	–	Mean Time Between Failures
MVP	–	Mission/Vehicle/Payload
NDRO	–	Non-Destructive Readout
NEMP	–	Nuclear Electromagnetic Pulse
NERVA	–	Nuclear Engine for Rocket Vehicle Application (Nuclear Engine)
NG&C	–	Navigation, Guidance and Control
NM/nm	–	Nautical Mile
O_2	–	Oxygen
OPGUID	–	Optimal Guidance Scheme

ABBREVIATIONS AND ACRONYMS (CONTINUED)

P	–	Pitch Axis
PCO	–	Program Controlled Output
PCM	–	Pulse Code Modulation
PM	–	Propulsion Module
PMD	–	Propellant and Maintenance Depot
PMF	–	Planar Magnetic Film
PW	–	Plated Wire
R	–	Roll Axis
RAM	–	Random Access Memory
RCS	–	Reaction Control System
RF	–	Radio Frequency
RNS	–	Reusable Nuclear Shuttle
S-IC	–	Saturn V First Stage
S-II	--	Saturn V Second Stage
S-IVB	–	Saturn V Third Stage
SCR	–	Scanning Laser Radar
SGLS	–	Space Ground Link System (Air Force)
SIU	–	Standard Interface Unit
SIRU	–	Strapdown Inertial Reference Unit
TBD	–	To Be Determined
TVC	–	Thrust Vector Control
USB	–	Unified S-Band
VHF	–	Very High Frequency
Y	–	Yaw Axis

1.0 INTRODUCTION

This document presents the results of a preliminary study performed to define an astrionic system(s) configured from "modular astrionics" which satisfies the requirements imposed by the broad spectrum of space tug missions.

The study is part of, and therefore generally consistent with, the objectives and study logic of the "Astrionic System Optimization and Modular Astrionics for NASA Mission after 1974" study, NAS8-14000, Mod. 158(revised. The objectives of this basic study are to:

- Perform a system engineering study of adaptation of the Saturn Instrument Unit Derivatives and propulsion stage astrionics for a family of potential NASA missions after 1974.

- Identify and classify functional astrionic requirements which are common, or peculiar, to boost, post-boost/rendezvous/docking/orbital/injection, and re-entry/recovery phases as applicable to each mission and vehicle combination.

- Establish the rationale for commonality of functions leading to definition of hardware and software modules for configuring minimum cost astrionic systems for each potential post 1974 launch vehicle/payload/mission combination.

- Conduct trade studies which substantiate the cost effectiveness of subsystems and components selected from instrument unit derivatives, or advanced technology, and include the effect existing and new or novel advanced management approaches have on overall cost.

- Identify "baseline" subsystems and components or identify additional study required to establish "baseline" elements for the "modular astrionics."

- Provide an RDT&E plan which includes preliminary specifications, test programs, schedules and costs for development and implementation of cost effective modular astrionic systems for each class or group of post 1974 launch vehicle/payload/mission combinations defined by the study.

The basic study flow logic is depicted in Figure 1-1. Each of the programs is a Mission/Vehicle/Payload (MVP) combination. Upon assignment for study, the mission and requirements of the astrionic system of the entire mission are evaluated and reported in a requirements specification. Other MVP's previously studied are related on a mission phase and astrionic system functional basis. Related requirements can at this point be challenged to minimize uniqueness, which can impose unnecessary impact on the layouts to follow.

Tradeoff analysis implements the requirements specification in hardware, software, and program elements. Each layout is a candidate for acceptance on the basis of cost effectiveness. Selection and recommendations of astrionic systems are acted upon by NASA to establish a baseline reference in the case of phase studies or for implementation in the case of changes to mature programs.

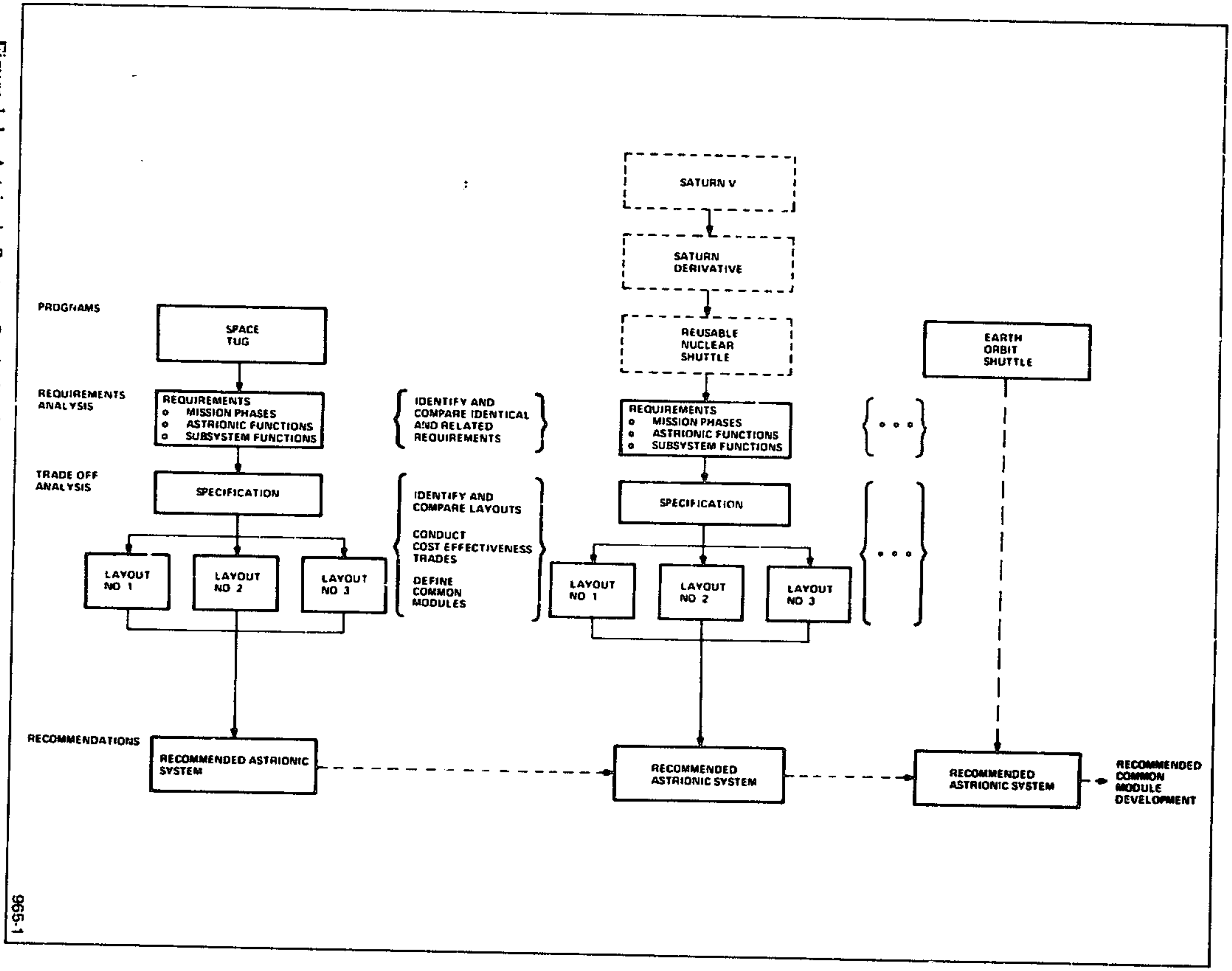

Figure 1-1. Astrionic System Optimization Study (Study Flow Logic)

The key to this analysis is the relating of one program to another to identify common requirements and to facilitate multilateral judgments across programs in the selection of system architecture, hardware, and programmatic impact.

As stated above, the study flow logic used for the tug astrionic system study is in general accord with this basic study logic. It is of significance to note, however, that one prime objective at this time was to define an astrionic system for the space tug in terms of weight, power, volume and other physical characteristics in a relatively short time frame. This requirement precluded the opportunity to perform detailed cost effectiveness trades and requirements analysis. Future study effort for space tug astrionics will use the data generated for this study as the departure point from which detailed requirements analysis and cost effectiveness trades can be made.

Section 2.0 of this report defines the space tug study groundrules and guidelines. Section 3.0 describes subsystem configurations and operations and summarizes the results of studies performed in the areas of maintainability, reliability, safety assurance, displays, radiation effects and orbital lifetime considerations at 100 nm. Section 4.0 summarizes the missions, describes astrionic system operation and details astrionic equipments in terms of usage, physical characteristics, installation, weight and power requirements. Section 5.0 discusses the astrionic system commonality between space tug and other space transportation systems, and Section 6.0 defines recommended future study effort. The appendices to the report present the details of the analyses performed to support the astrionic system(s) defined.

A space tug progress report (Appendix C to "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974," Progress Report – 16 April 1970 to 15 June 1970, IBM No. 69-K44-0006F, 26 June 1970) was issued in June. That report defined the status of the tug study as of that date and described some of the preliminary hardware and concepts being considered for the astrionic system. Since that time, additional analysis has been performed to further define the astrionic system(s). This document represents the results of the study to date and therefore supercedes and replaces, in its entirety, Appendix C to IBM Report No. 69-K44-0006F.

2.0 STUDY GROUNDRULES AND GUIDELINES

2.1 GENERAL SPACE TUG GROUNDRULES

These groundrules and assumptions form the background for this study.

- The space tug (see Figure 2-1 for baseline configuration) includes four basic modules (crew, propulsion, astrionic, and cargo) and auxiliary kits for special-purpose missions (landing legs, manipulation arms, etc.) as required. Combinations of the modules and kits will be used for each mission.

- The tug is based and maintained in space and on the ground.

- The tug may be configured for manned or unmanned missions. In the unmanned configuration, the tug will be operated automatically or by remote control from the earth or from other space elements.

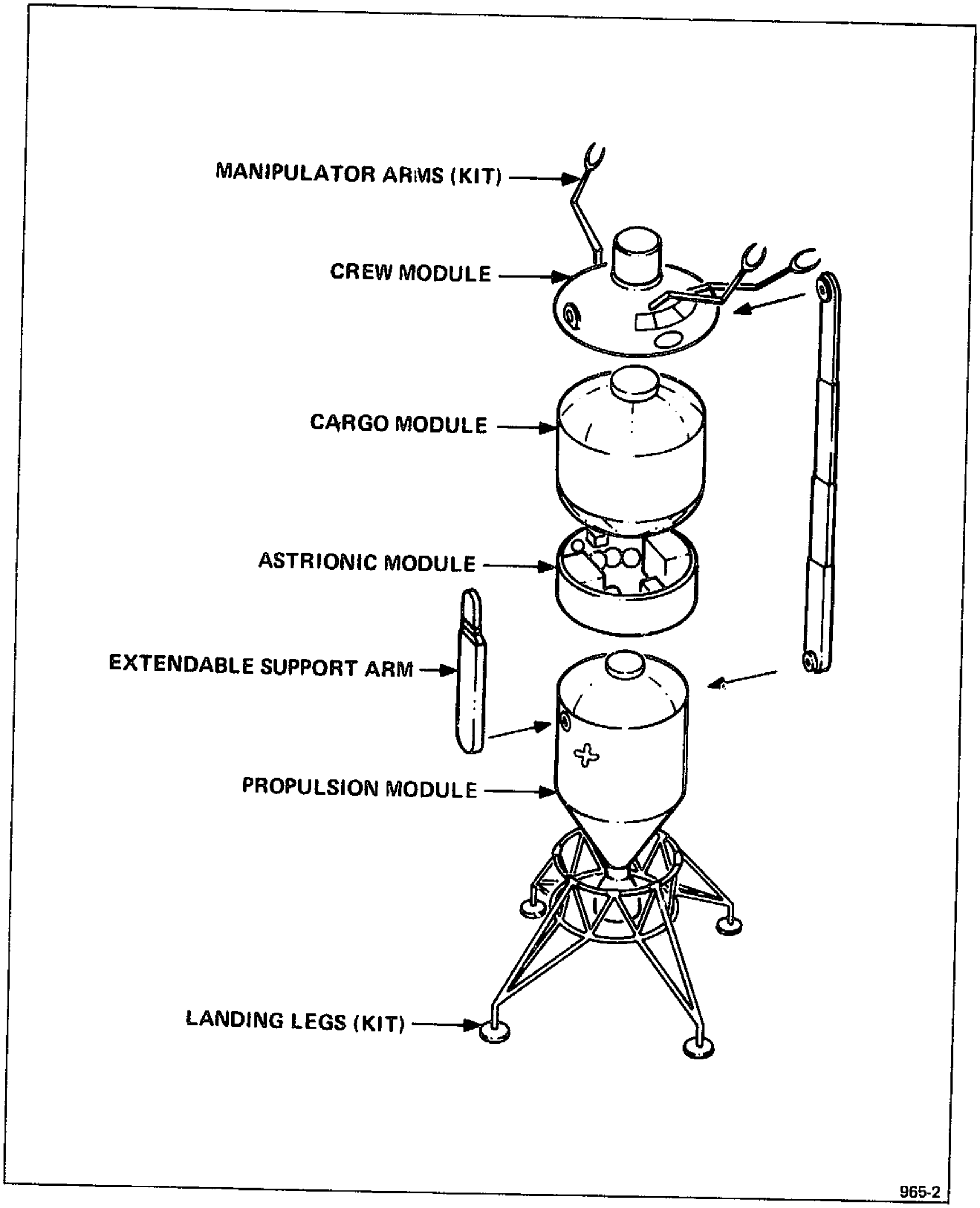

Figure 2-1. Space Tug Baseline Configuration

- The space tug will be delivered to orbit by a space shuttle or Saturn derivative vehicle.

- The lunar orbit space station is assumed to be in a polar orbit at a circular altitude of 60 nautical miles.

- The space tug shall be compatible with the earth and lunar orbiting space stations, the space shuttle, and the reusable nuclear shuttle (RNS).

- The space tug astrionic module design shall minimize the need for ground support.

- The space tug shall be capable of maintaining a quiescent status for up to 180 days in earth or lunar orbit when docked to other space elements or free-flying. Quiescent periods of up to 42 days will be required on lunar surface (14 or 28 days + 14 days contingency).

- The space tug shall be capable of going from the quiescent state to a fully operational state within two hours.

- The reusable space tug shall have a minimum lifetime goal of ten years and the capability of being reused at least ten times by replenishment of consumables and through performance of required maintenance. Maintenance, as required, and reconfiguration capability while in space residence is considered mandatory. Major refurbishment may necessitate the need for the tug to be returned to earth.

- Maximum crew safety and a high probability of fulfilling all space tug functions and objectives shall be a design goal. Subsystems identified as necessary for crew survival will be designed such that no single failure or credible combination of failures will result in loss of life.

- In the manned mode, the space tug can be piloted by one crewman.

- The crew module will serve as the primary crew living quarters and a base of operations (mission control) for manned missions.

- The crew module will contain an airlock for EVA and multi-simultaneous EVA operation capabilities.

- Tug communication systems will be compatible with the Manned Space Flight Network, Deep Space Network, SGLS (DOD), available Communication Satellite Systems and with space elements, such as the space stations, shuttles, etc., depending on the mission.

- The space tug attitude reference system shall have complete freedom in all axes.

- The space tug shall have neuter docking devices compatible with all space vehicle hardware elements.

- Minimum interfaces are required between the space tug and its payload to reduce complexity and increase the flexibility of the kinds of payloads to be transported. However, consideration should be given to how the space tug communications and power subsystems could support the payload.

- The primary propulsion system will be LOX/LH_2. Secondary propulsion systems may also be LOX/LH_2.

- The first operational flight will be flown no earlier than 1977.

- The tug astrionic system will have the capability to automatically control and monitor the refueling process with automatic shutoff and disconnect capability.

- The sync orbit mission of the tug for the IBM study will be a baseline.

- Only cooperative satellites are considered for all retrieval missions.

- More than one low-energy mission may be performed between refuelings.

- The space tug maximum diameter must be less than 15 feet; one throttleable and gimballed engine will be used.

- The space tug astrionic system shall not be constrained by lunar lighting conditions. The tug shall be capable of landing at all lunar latitudes and longitudes with appropriate time-phasing.

2.2 ASTRIONIC SYSTEM GROUNDRULES AND GUIDELINES

- Initial study activity of the tug astrionic system will include all elements necessary for space tug attitude control, engine(s) control, guidance, navigation, communications, checkout, sequencing, environmental survival of astrionic equipment, data management and reporting, safety assurance, signal distribution, astrionic power generation and distribution, and the crew display and command system.

- The astrionic system shall be self-sustaining.

- The thermal control subsystem will be sized only to support the astrionic module.

- The power subsystem will be sized for the total astrionic system with a quick look at an integrated power system for the total tug.

- The RCS engine controls and main engine actuators are considered as part of the hardware for other modules. The astrionic module will supply and accept the signals to operate this equipment.

- Power and environmental conditioning may be provided to the space tug astrionics by a space element when the tug is docked with the element.

- The astrionic module will be packaged to allow a remove and replace maintenance and reconfiguration concept in space.

- As a minimum, equipment will be designed for a fail safe and repair operation with a fail operational/fail safe condition on equipment required for crew safety.

- The astrionic module will be nominally 14 feet in diameter, and height will be optimized.

- The astrionics to support crew displays and command will be sized for maximum of three crew stations in the crew module.

- The design goal of checkout, either onboard or from an external source, will be to locate failures to a lowest replaceable unit (LRU) consistent with the philosophy of modular astrionics.

- Crew module backup systems required after separation from the astrionic module are not considered in this study.

- The astrionic system will be capable of checkout and monitoring for the total tug consistent with the tug maintenance concept.

- A standard physical interface for astrionic equipment and a standard electrical interface for all electrical equipment will be provided to simplify interchange of components and data transfer.

- The astrionic module will be of modular design providing the capability of automatic operation, manual control, and remote operations depending on the representative missions defined in the mission plan.

- Between missions, maintenance will be performed as required to upgrade the reliability to the required level. During the mission, maintenance shall be limited to switching-in redundant systems.

- The low earth orbit is defined as between 80 and 280 nautical miles with a range of inclination from 28.5 to 55°. The lunar orbit is assumed to be a polar orbit at a circular altitude of 60 nautical miles.

2.3 DEFINITIONS

- Remote operation – The control of the system is accomplished from an external source.

- Automatic operation – The system is completely independent of any external control or astronauts.

- Cooperative satellites – Satellites that are attitude stabilized, have corner reflectors, and have a docking adapter(s).

3.0 SUBSYSTEM DESCRIPTION

This section of the report describes the astrionic subsystems and the components of the subsystems required for the space tug astrionic module. A summary of the space tug design mission descriptions and the functional requirements are presented at the first of this section. This is followed by the subsystem component descriptions and summaries of other studies related to the design of the space tug astrionic module.

3.1 MISSION DESCRIPTIONS

Extensive effort has been applied to this area to define the typical spectrum of missions to be performed by the space tug. This effort was necessary because definitive mission descriptions did not exist at the beginning of this study.

The approach to the mission analysis was to define the total spectrum of probable tug missions and to group these into similar types of missions. A design mission, which would define the most stringent astrionic requirements for that type of mission, was then selected for each group. These design missions were used as the basis for the space tug modular astrionics study. The following is a brief summary of the design missions used for the astrionic system requirements analysis. The space tug design mission profiles are summarized in Figure 3-1. A more detailed mission description and analysis for each of these missions is included in Appendix A.

3.1.1 Synchronous Orbit Mission

3.1.1.1 Reusable Stages

This mission will use two unmanned reusable space tugs (propulsion module (PM) and astrionic module) to place a payload in a synchronous equatorial orbit and return another payload to low earth orbit.

The space tugs are brought to a 100 nautical mile (nm) circular orbit at an inclination of 28.5° by the space shuttle and configured for the mission. The first tug will perform a partial burn to start the payload on its journey. It will then return to low earth orbit. The second tug will complete a Hohmann transfer to synchronous orbit, jettison the payload, maneuver to and pick up another payload and perform another Hohmann transfer back to low earth orbit. The space tugs will then be deactivated and left in low earth orbit until the next mission or returned to earth by the space shuttle.

3.1.1.2 Expendable Stage

This mission will use one expendable space tug (PM and astrionic module) to place a payload in a synchronous equatorial orbit.

The space tug will be brought to a 100 nm circular orbit at a 28.5° inclination by the space shuttle and configured for the mission. A two-burn Hohmann transfer burn will place the payload in a synchronous orbit. The space tug will jettison the payload and perform a depletion burn to send the PM and astrionic module into space.

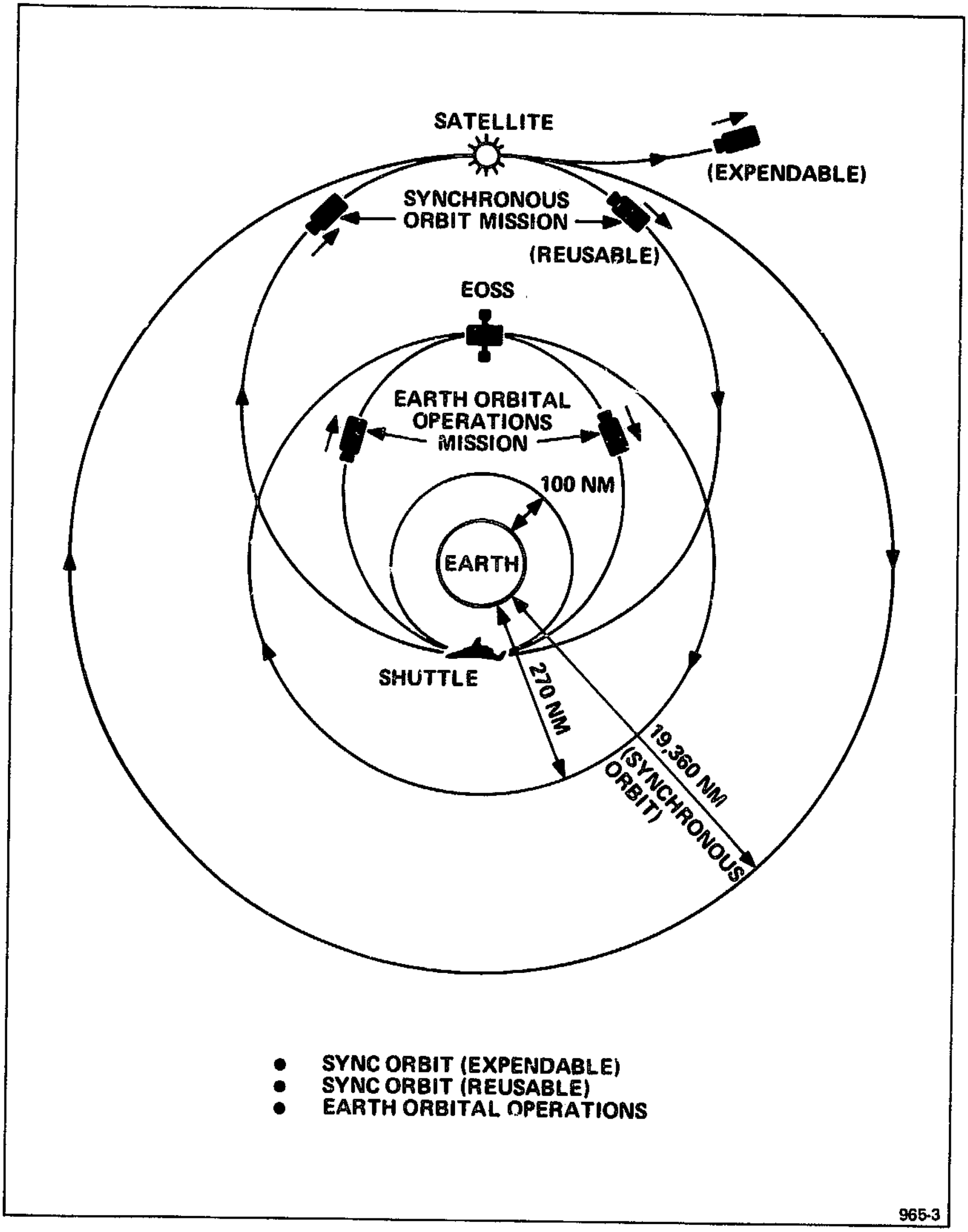

Figure 3-1. Space Tug Design Mission Profiles (Sheet 1 of 2)

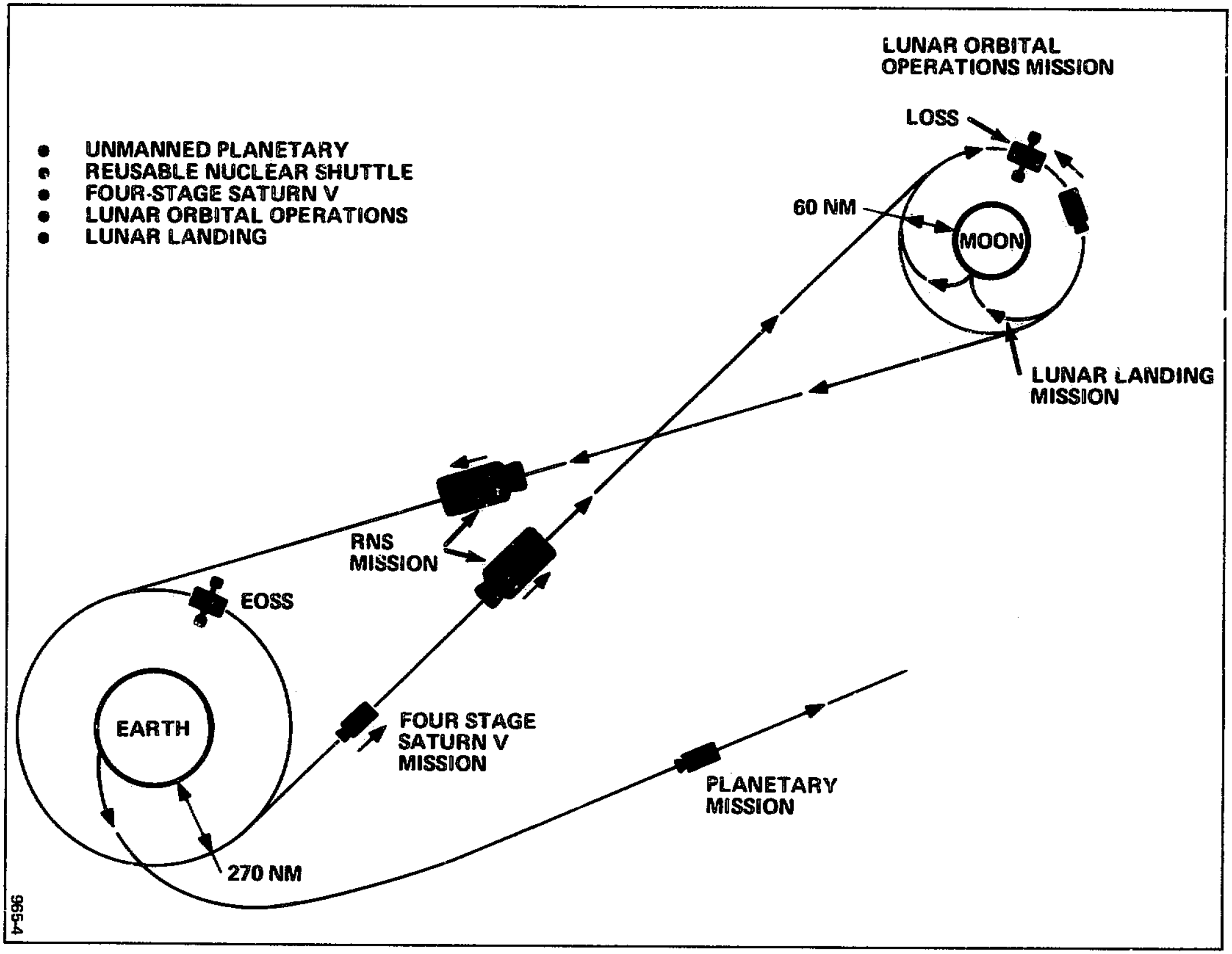

Figure 3-1. Space Tug Design Mission Profiles (Sheet 2 of 2)

3.1.2 Earth Orbital Operations Mission

This mission will use a manned or unmanned reusable space tug for orbital operations in conjunction with a space station in earth orbit.

The mission will begin with the space tug inactive and docked to the space station in earth orbit. The tug will be activated and will undock from the space station. The space tug will then perform maneuvers to transfer (possibly with a payload) from a 270 nm to a 100 nm orbit, will pick up a payload delivered to earth orbit by a space shuttle or a Saturn derivative vehicle, and will perform the maneuvers to return to the space station at a 270 nm altitude. The space tug will then be deactivated until another active mission is required.

3.1.3 Lunar Orbital Operations Mission

This mission will use a manned or unmanned reusable space tug for orbital operations in conjunction with a space station in lunar orbit. This mission has essentially the same requirements as the earth orbital operations mission, except that the lunar orbiting space station is at a 60 nm altitude.

3.1.4 Unmanned Planetary Mission

This mission will use one reusable space tug (PM and astrionic module) and one expendable space tug (PM and astrionic module) to boost a planetary payload on a planetary trajectory.

This mission will begin with the space tugs configured for the mission and docked to the earth orbiting space station. The tugs will be activated and will undock from the space station. A partial burn of the first tug will start the payload on its journey. The tug will then return and dock to the space station. The second tug will ignite at first tug cutoff to give the payload the required delta V for the planetary mission. The second tug will then be jettisoned into space.

3.1.5 Reusable Nuclear Shuttle Earth/Moon Mission

This mission will use a reusable nuclear shuttle (RNS) with a space tug astrionic system to transfer payloads between earth and lunar orbits.

This mission begins with the RNS deactivated and docked to a propellant and maintenance depot (PMD) in earth orbit. The RNS will be activated and maneuver a safe distance from the space station. The RNS will ignite and transfer the payload to a lunar orbit in the vicinity of a lunar orbit space station (LOSS). After the payload is removed, the RNS will ignite and return to earth orbit. The RNS will be docked to the PMD and deactivated until required for another mission.

3.1.6 Lunar Landing Mission

This mission will use a fully configured space tug for the landing of a payload on the lunar surface and return of men to the LOSS.

This mission will begin with the space tug configured for the mission, deactivated, and docked to the LOSS. The tug will be activated and will undock from the LOSS. The tug will perform a series of burns to land at a designated spot on the lunar surface. The space tug element not being used during the lunar surface stay will then be deactivated.

After a 14 or 28 day nominal stay, the tug will be activated and will burn to achieve orbit and return to the LOSS. After docking, the tug will be deactivated until required for future use.

3.1.7 Four Stage Saturn V Mission

This mission will use a Saturn V vehicle (without IU) with a propulsion module and astrionic module as the fourth stage. This configuration will be used to transfer payloads from earth to lunar orbit.

The configuration will be checked out prior to launch. Complete S-IC and S-II and a partial S-IVB burn will place the payload in earth orbit. A second burn to completion of the S-IVB and a partial tug burn will place the payload on a lunar trajectory. After 3 to 4 days of translunar coast, the tug will transfer into lunar orbit and perform a gross rendezvous with the LOSS. After the payload is deactivated, this mission is complete. The tug may, however, be used for other missions after deactivation.

3.2 ASTRIONIC SYSTEM PRELIMINARY FUNCTIONAL REQUIREMENTS ANALYSIS

The analysis of the space tug functional requirements is presented in Appendix B to this report. The preliminary functional requirements have been identified for the lunar landing and synchronous orbit missions, with requirements relating to other tug missions also identified. The effort has concentrated mainly on the navigation, guidance and control requirements. These preliminary requirements were used as a basis for selection of equipment for the space tug astrionic system.

3.3 DATA MANAGEMENT SUBSYSTEM DESCRIPTION

The data management subsystem integrates the tug astrionics by providing for all subsystem-to-subsystem signal flow. In addition, the data management subsystem provides the computational support for the following astrionic functions:

- Attitude control (TVC or RCS)
- Engine Thrust Level Control
- Guidance
- Navigation
- Checkout and Hardware Reconfiguration Control
- Data Monitoring and Telemetry
- Sequencing

- Data Bus Control
- Software Management
- Display Support
- IMU Processing
- Maintenance Support
- Refueling Support

To perform the above functions, the data management subsystem will consist of the following hardware:

- A central processing unit (CPU) of a medium speed range (300K to 400K ops/sec)
- 32K to 64K of random access main memory
- A bus control and input/output unit
- A configuration assignment unit (CAU) for switching units when a failure occurs.
- A magnetic tape unit for mass storage.
- A 32K display memory for storing display skeletons.
- An auxiliary monitor computer.

The following is a description of each of the hardware components and their uses. A schematic of the data management subsystem and its interfaces is shown in Figure 3-2. Appendix C of this report gives a more detailed analysis of the selection of the hardware and its operation.

3.3.1 <u>Central Processing Unit</u>

A CPU in the medium speed range (300K to 400K ops/sec) was selected to perform the data processing and data management and control required for the space tug missions. The CPU will be a parallel machine with floating point operation and will have a 128 instruction repertoire. It will be designed for use on all space tug missions, with the number used depending on the complexity of the mission.

3.3.2 <u>Random Access Main Memory</u>

A random access memory will be used for the space tug memory requirements. It will probably be a monolithic memory and each astrionic module will have either 32K or 64K of main storage depending on the mission. The memory would use several basic operating memories (BOM) each containing single bit words. Each BOM would contain 32K x 1 bit words. The memory would contain 32 BOMs for data storage and 7 BOMs for error correction coding, or a total of 32K x 39 bit words.

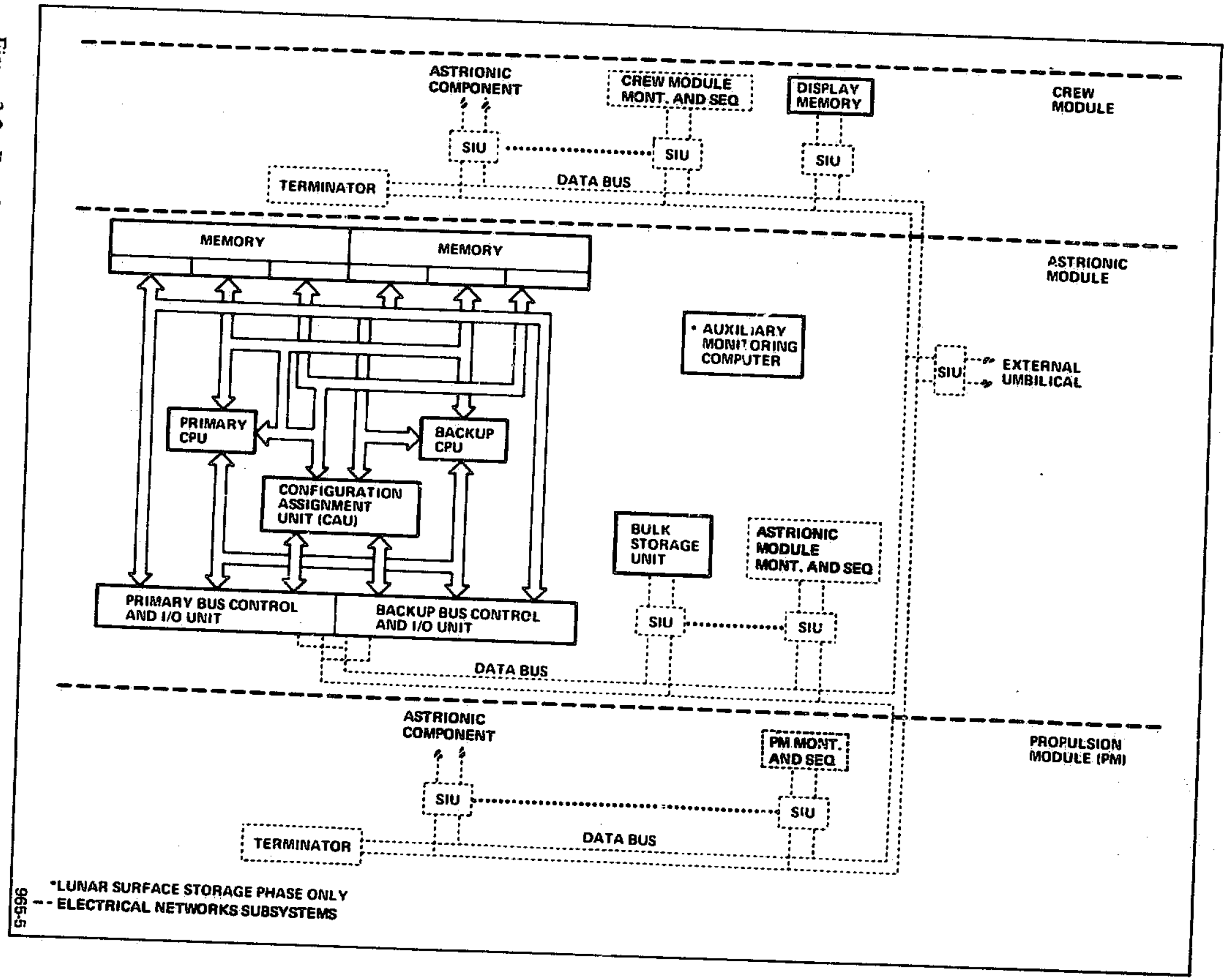

Figure 3-2. Data Management Subsystem and Interfaces

3.3.3 Bus Control and I/O Unit

The bus control and I/O unit will serve to control all information flow between the CPU and the data bus and between the random access memory and the data bus. It will send and receive address and data information to and from the data bus in serial digital form for use by the CPU and main memory. It will also perform the same operations between the CPU and the main memory.

3.3.4 Configuration Assignment Unit

A configuration assignment unit (CAU) is required to monitor itself, the main memory, the CPU, and the bus control units. When a failure occurs in any of these primary units, the CAU will detect the failure in the primary unit and switch to a backup unit.

3.3.5 Magnetic Tape Unit

A magnetic tape unit was selected for the mass storage requirements for the space tug missions. The unit will store redundant programs, extensive checkout and diagnostic programs and programs for refueling support, as well as any mission data which may require mass storage.

3.3.6 Display Memory

Memory is required to store display skeletons (for manned missions). The display skeletons are a bulk storage requirement requiring fast access time. Since the magnetic tape unit is relatively slow, the display memory was added for the manned space tug missions. The monolithic or magnetic memory will have 32K of 32-bit words and may be "read only" if the number and type of skeletons are fixed for a given vehicle.

3.3.7 Auxiliary Monitor Computer

An auxiliary monitor computer was added for the storage phase of the space tug on the lunar surface. The unit will be a small general purpose computer which will support monitoring and displaying of critical parameters to the crew during the 14, 28 or 42 day stay on the lunar surface. This will allow the data management hardware to be powered down during this period to allow reliability to be maintained at a high level for the entire mission.

3.4 NAVIGATION, GUIDANCE AND CONTROL SUBSYSTEM DESCRIPTION

The purpose of the navigation, guidance and control subsystem is to navigate, guide and control the space tug in performing its desired mission. The navigation function will determine the position and velocity of the vehicle from measurements made on-board the vehicle. The guidance function will compute the maneuvers necessary to achieve the desired end conditions of a trajectory. The control function will perform the maneuvers determined from the guidance function. Figure 3-3 is a summary schematic of the recommended navigation, guidance and control subsystem.

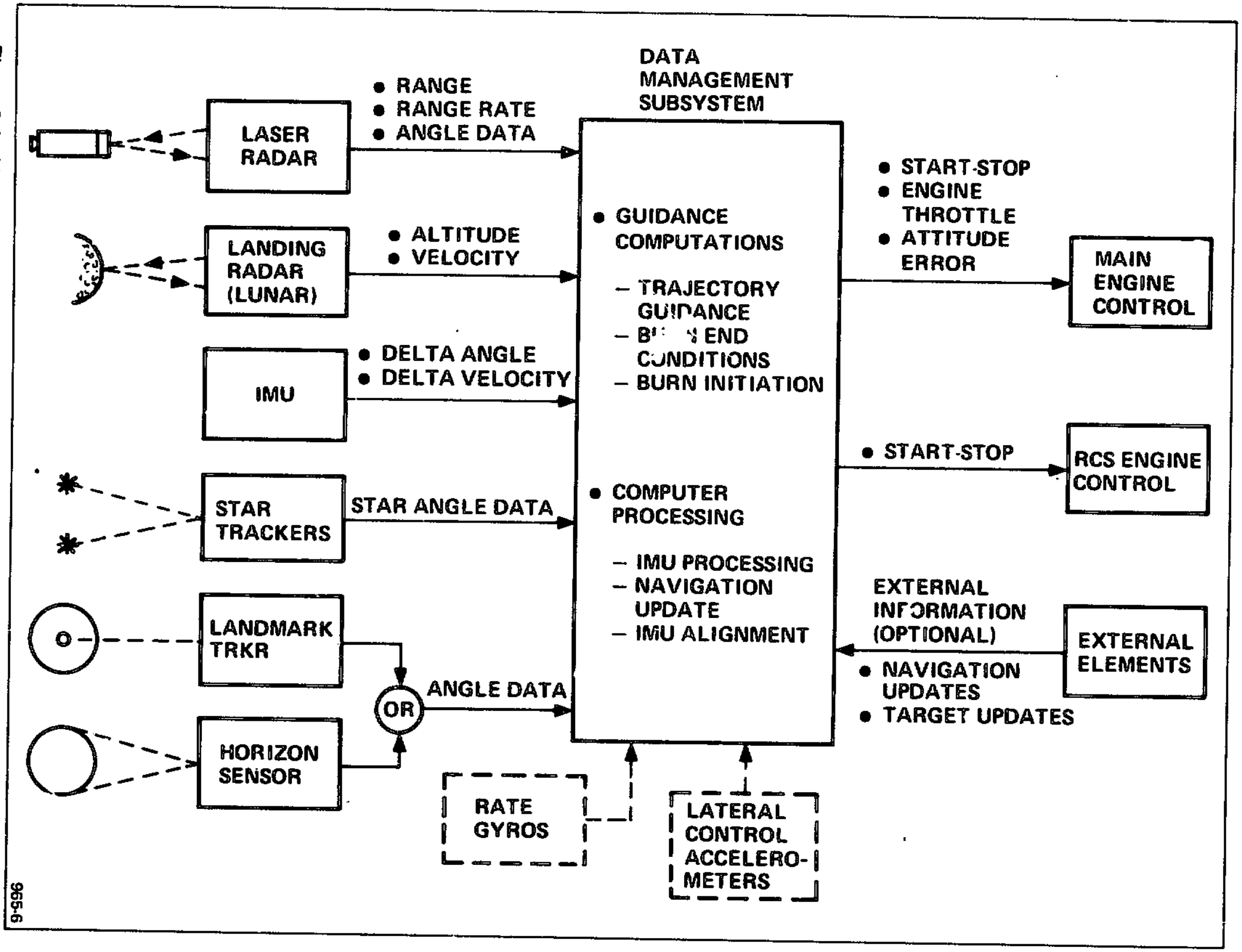

Figure 3-3. Navigation, Guidance and Control Subsystem

3.4.1 Navigation

To perform the above functions, the following hardware is used to fulfill the navigation functional requirements for the space tug:

- IMU (Hexad Strapdown)
- Laser Radar
- Star Trackers (2)
- Landmark Tracker
- Horizon Sensor
- Landing Radar

The following is a description of each of the pieces of hardware and their uses. A more detailed analysis of requirements and the selection of the hardware and its uses is given in Appendix D of this report.

3.4.1.1 IMU

An IMU is required for active mission phases to measure vehicle thrust acceleration and provide vehicle attitude reference. A strapdown IMU with six gyros and six accelerometers (hexad configuration) whose input axes are colinear with the normals to the face of a regular dodecahedron was selected. The IMU will be operated in a classical pulse-torque-restrained strapdown control mode, with IMU outputs fed to the CPU for processing. The hexad configuration will allow failure detection and isolation to any two gyro or accelerometer axes and detection of a third failure.

3.4.1.2 Laser Radar

The purpose of the laser radar is to provide range, range rate, angular position and angular rate with respect to a target vehicle or location for automatic rendezvous and docking. It can also be used as a landing aid for a lunar landing at a prepared site.

The laser radar selected is being developed for NASA by ITT. The radar consists of an electronic unit and a sensor unit. The ungimballed sensor will scan a field-of-view of 30° by 30° and acquire and track targets to a range of 75 nautical miles. Corner cube reflectors, either active or inactive, mounted on the target are used with the laser radar. The data from the laser radar is input to the central computer for processing and utilization.

3.4.1.3 Star Tracker

Two star trackers were selected for automatic IMU alignment. Although two star trackers are not mandatory for alignment, the two star trackers operating simultaneously provide greater accuracy in alignment and allow redundancy for degraded accuracy alignments with failure of one unit. The star trackers, in conjunction with either a landmark tracker or horizon sensor, are also used for navigation updates during the space tug missions.

The star trackers consist of two units; a sensor subassembly and an electronics subassembly. The sensor unit is mounted with external access to the stars. Star position information stored in the computer is used to initially point the sensor unit. When the selected stars are detected, the electronics unit converts the error signal from the sighting to digital form and sends it to the computer. The computer uses this data to update the pointing commands in order to zero the error.

3.4.1.4 Landmark Tracker

The landmark tracker is used in conjunction with the star trackers for autonomous vehicle state updates. It is used for distances up to a maximum of 10,000 nm from a celestial body.

The landmark tracker consists of a sensor unit and an electronic unit. The sensor unit, consisting of a vidicon and optics, is gimballed in the pitch and roll axes and mounted external to the vehicle. The landmark tracker sensor will detect landmarks, such as islands, and compare data from several sightings with information previously stored in the computer. Navigation errors are defined in this process and are used for navigation updates for the space tug.

3.4.1.5 Horizon Sensor

The horizon sensor is used in conjunction with the star trackers for autonomous navigation updates beyond an altitude of 10,000 nm. The horizon sensor consists of a sensor unit and an electronic unit. The sensor unit utilizes four infrared search-track units to track the celestial body horizon in four planes separated 90 degrees in azimuth. The sensor unit, mounted external to the vehicle, detects the infrared horizon of a celestial body and inputs this information to the central computer for processing of navigation updates.

3.4.1.6 Landing Radar

A lunar landing radar, used in conjunction with a laser radar, is used to provide velocity and altitude with respect to the lunar surface during a lunar landing. It also supplements the IMU inertially measured quantities during the lunar landing.

The landing radar is composed of an antenna assembly, an electronic assembly and a control assembly. The antenna is deployable and provides accurate information for the lunar landing from 25,000 feet to touchdown. Velocity components from the landing radar are input to the central computer processing for use during the descent and ascent phases of the lunar landing mission.

3.4.2 Guidance

The guidance analysis has not identified requirements for any hardware above that required for navigation and other functions or subsystems. Guidance schemes were analyzed to determine the schemes best suited to the space tug guidance function requirements.

An optimal guidance (OPGUID) scheme was selected for rendezvous, high energy maneuvers, lunar ascent/descent and lunar orbit insertion plane change maneuvers to be used during the spectrum of space tug missions. The docking/undocking inputs are received from the rendezvous radar. The OPGUID scheme is a general purpose scheme which can be used for all space tug design mission phases. A more detailed discussion of the guidance schemes considered and information on operation of the schemes are found in Appendix E to this report.

3.4.3 Control

The control analysis has evaluated the control schemes and laws which would be required to implement translational and attitude maneuvers using reaction control system (RCS) and main engine thrust vector control (TVC) subsystems. The schemes used for controlling the vehicle were then used to estimate the software requirements dictated by the control function. A detailed discussion of the control analysis is given in Appendix F to this report.

Except for the four stage Saturn V mission, the control analysis has not identified requirements for any hardware above that required for navigation and other functions or subsystems. Possible additional hardware for the four stage Saturn V mission may include:

- 3-axis rate gyro package
- 2-axis lateral accelerometer package

3.4.3.1 Rate Gyro Package

The rate gyro package may be added near the four stage vehicle's point of maximum stability or minimum bending. The package would provide data from three perpendicular axes to enhance the vehicle stability under marginal flight conditions.

3.4.3.2 Accelerometer Package

The accelerometer package may be used to provide load relief for lower stages during the boost-to-orbit phase of the four stage Saturn V mission. The package would provide lateral accelerations in the pitch and yaw axes.

3.5 ELECTRICAL POWER SUBSYSTEM DESCRIPTION

The electrical power subsystem generates the electrical power necessary to operate the onboard electronic and electromechanical components.

The following astrionic equipment is required for the electrical power subsystem:

- 2 Kw Fuel Cell(s)
- Battery
- DC Regulator
- Battery Charger

The following gives the description and uses of these items. A more detailed description of the equipment operation and associated trade studies is given in Appendix G.

3.5.1 Fuel Cell

A two kilowatt fuel cell was selected as the basic component for the electrical power subsystem. The fuel cell will generate power required for the astrionic equipment. It uses liquid hydrogen and liquid oxygen as reactants to generate the power, with water as a by-product of the power generation.

The amount of reactant required is dependent upon the tug power requirements and length of the mission and thus is mission dependent.

Likewise, the size of the H_2 and O_2 tanks required to store the reactants is dependent on the mission requirements.

3.5.2 Battery

A battery is required for the manned space tug missions for emergency power for critical components if the fuel cells failed during a mission. The battery selected is a 28 volt dc power supply with a 440 amp-hour nominal capacity. A smaller battery may be used for peak loading for the reusable nuclear shuttle mission.

3.5.3 DC Regulator

DC regulators are required to provide a very accurate voltage reference for telemetry calibration, signal conditioning and sensor applications.

3.5.4 Battery Charger

The battery charger is connected between the fuel cells and the battery. It is used to recharge the reusable battery using power outputs from the fuel cell configuration.

3.6 ELECTRICAL NETWORK SUBSYSTEM DESCRIPTION

The electrical network subsystem will consist of the circuits and equipment required to distribute power and signals to the space tug electrical equipment.

The following hardware has been identified for inclusion in the electrical network subsystem:

- Standard Interface Units (SIU)
- Data Bus
- Monitoring and Auxiliary Monitoring units
- Power Distributor
- Auxiliary Power Distributor

- Junction Boxes
- Wires and Cables

This equipment and its use are discussed in the following sections. A more detailed description of the trades concerning the uses of this equipment is given in Appendices C (Items 1, 2, and 3), and G (Items 4 through 7).

3.6.1 Standard Interface Units

The standard interface units (SIU) are used to adapt the input/output unit of the space tug equipment requiring interface with the data management subsystem with the data bus. The SIU is identical for all monitored units.

The SIUs contain the circuitry necessary to decode the digital address of the measurement or command, to receive the command or data from the data bus or to deliver data to the data bus. The SIU converts the data to serial or parallel and analog or digital, depending on the form required by the data bus or the device.

3.6.2 Data Bus

The data bus concept uses (tentatively in the design) two pairs of twisted wires terminated at the end of the bus acting like a transmission line. These wires, called the data line, are used for two-way digital data transfer. The data bus is controlled by the bus control and input/output unit and transfers digital data between the bus control unit and the standard interface units.

3.6.3 Monitoring and Auxiliary Monitoring Units

The monitoring units are used to monitor parameters and sequence equipment not accessed by other standard interface units. The monitoring units will be SIU and I/O devices which multiplex measurements, hardwired from measuring instrumentation, and provide a path for sequence command to electronic switching devices. The SIU and I/O address logic will handle the sequencing commands.

The auxiliary monitoring units are identical to the monitoring units, but are used to monitor critical parameters during the storage phase on the lunar surface for the lunar landing mission.

3.6.4 Power Distributor and Auxiliary Power Distributor

A main power distributor is used to distribute power from the fuel cell(s). The auxiliary power distributors, in turn, supply the high and low current auxiliary power for load distribution of the astrionic subsystem equipment.

3.6.5 Junction Box

Junction boxes are provided on each of the eight component mounting panels of the astrionic module to distribute power and signals to the equipment mounted on each panel.

3.6.6 Wires and Cables

The wires and cables are an estimate of the wires required to distribute the electrical power for the space tug astrionic system.

3.7 COMMAND AND CONTROL SUBSYSTEM DESCRIPTION

The space tug command and control subsystem will consist of the command, telemetry, tracking and voice communications external to the space tug vehicle.

The command and control subsystem will use the following hardware:

- Unified S-Band (USB) Equipment
- USB Diplexer, Antenna Switch and Power Divider
- USB Omni Antennas
- USB Hi-Gain Antenna (Transponder)
- USB Hi-Gain Antenna Control
- VHF Transceiver Equipment
- VHF Diplexer and Power Divider
- VHF Omni Antennas
- VHF Command Receiver
- Command Decode Electronics
- TV Camera
- TV Camera Control
- Audio Equipment

A block diagram of the command and control subsystem is shown in Figure 3-4. The following is a description of each of the hardware items for the command and control subsystem. A more detailed analysis of the items in this subsystem is given in Appendix H to this report.

3.7.1 USB Equipment

The USB equipment was selected to perform the unified requirements for command, tracking, telemetry and voice. Therefore, it is the prime link between the space tug and ground control stations. The frequency range is 2.1 to 2.3 GHz.

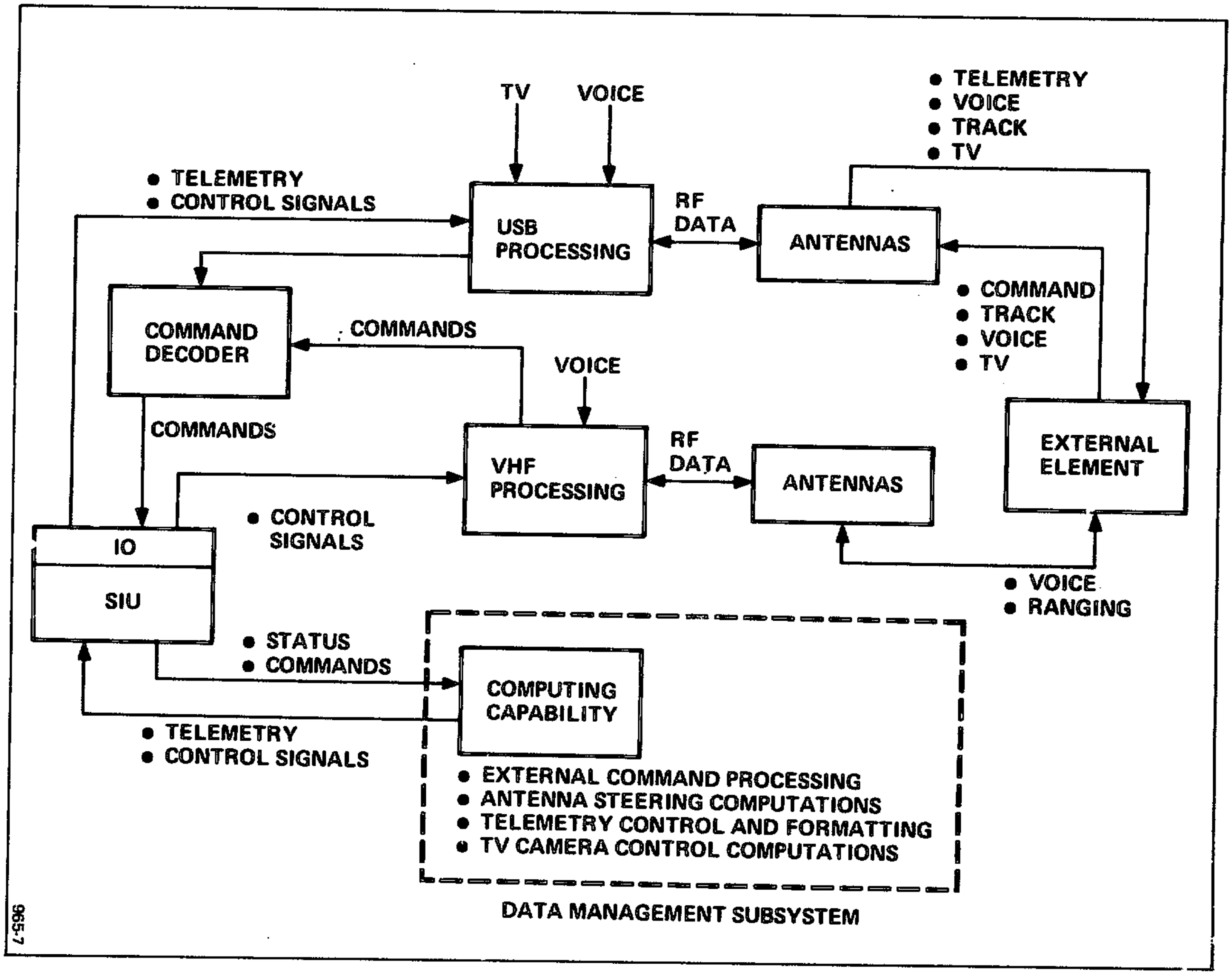

Figure 3-4. Command and Control Subsystem

The USB equipment consists of a transmitter, a power amplifier, a receiver with preamplifier, a transponder for turn-around of the ranging signal, a premodulator processor providing subcarrier and baseband makeup, and the demodulators for the uplink signals. This equipment processes incoming signals (such as command or tracking) and outgoing signals (such as telemetry, TV and voice).

3.7.2 USB Diplexer, Antenna Switch and Power Divider

This equipment is used to provide a path for the command and control links between the USB equipment (transmitter and receiver) and the antennas. The USB diplexer receives data from the USB equipment, sends it into the USB antenna switch (which determines which antenna(s) should be operating) and then into the USB power divider which directs the data to the appropriate operating omni antennas.

3.7.3 USB Omni Antennas

Four USB omni antennas are required for complete omni-directional coverage for space tug command and control. These antennas are flush-mounted types and are connected through the power divider.

3.7.4 USB Hi-Gain Antenna

The USB hi-gain antenna is required on the space tug for long-range communications and high-bit rates when using TV. The hi-gain antenna has tentatively been identified as a steerable, deployable four-foot diameter parabola with a gain of approximately 26 db. The antenna is deployed through the door of the astrionic module. A gain switching feature is included on the antenna.

3.7.5 USB Hi-Gain Antenna Control

A USB hi-gain antenna control capability has been included on the space tug to deploy and steer the hi-gain antenna. The antenna control will receive steering commands from the data management subsystem in body coordinates, thus directing the antenna to its appropriate target.

3.7.6 VHF Transceiver Equipment

The VHF command and control equipment will be the prime voice link between the space tug and other space elements, including EVA, and will be a backup to the USB system for voice and low data rates. The VHF equipment consists of a transmitter, a receiver, a modulator and demodulator and a ranging turn-around capability. This equipment will be used for manned missions.

3.7.7 VHF Diplexer and Power Divider

The VHF diplexer is used to combine signals from the transmitter and to the receiver. These signals are transferred to the power divider, which directs the data to the appropriate omni antenna.

3.7.8 VHF Omni Antennas

Two VHF omni antennas are required for omni-directional coverage for communications. The omni antennas are blade type antennas which extend a few inches from the skin surface. A two watt transmitter appears sufficient for 100 nm vehicle-to-vehicle communications.

3.7.9 VHF Command Receiver

The VHF command receiver is a small, lightweight, low power drain device to receive commands to power the astrionic module during earth storage phases (such as 180 day free drift). The receiver will operate in the 136-148 MHz band. Two units at separate frequencies will be used. The VHF omni antennas are used to receive the signals.

3.7.10 Command Decode Electronics

The command decode electronics performs the functions of a sub-bit decoder and to interface with the standard interface unit to put uplink commands on the data bus. The decode electronics will decode, verify and command driver circuits for command decoding during active mission phases and to bring the quiescent astrionic module out of the storage mode.

3.7.11 TV Camera

The TV camera may be located in the astrionic module for unmanned missions or the crew module for manned missions. The camera will be a commercial grade black and white TV requiring 2.9 MHz bandwidth (color will require 4.2 MHz bandwidth).

3.7.12 TV Camera Control

The TV camera control unit will provide for remote control through the command link for lens and filter change, zoom control and complete articulation for the camera.

3.7.13 Audio Equipment

The audio equipment is used for manned missions and provides the line drivers and interface conditioning between the intercoms (head sets) and the modulators/demodulators of the command and control equipment.

3.8 STRUCTURES DESCRIPTION

The space tug astrionic module structure provides the load-bearing member for the astrionic module and provides a base for mounting equipment on the astrionic module. It will also aid in micrometeoroid protection. An astrionic module ring was groundruled for the study to provide the flexibility of diverse configurations for the tug for various missions requirements.

An aluminum honeycomb sandwich shell was one structural option which was studied. (See Figure 3-5.) This shell is 14 feet in diameter and 4 feet high, with a thickness envelope of less than 1 inch. The radiators are connected to the outside of the shell with low conductivity standoff connectors. The component mounting panels are connected to the interior of the shell. The shell structure exhibited the lightest structural weight of the options studied.

Another option for the astrionic module is an open frame structure. (See Figure 3-6.) With this option, eight load bearing columns (posts) in the structure are connected at the top and bottom by circular structural rings. The open frames allow the radiator/component mounting panel combinations to be hinged or removable from the openings in the structure. For this structure, the component mounting panels may be hinged to swing either inward or outward with components mounted internally, externally or both. These options would allow maximum accessibility for maintenance.

The results of the analysis for structural and packaging concepts are given in Appendix I to this report.

3.9 THERMAL CONDITIONING SUBSYSTEM DESCRIPTION

The thermal conditioning subsystem provides temperature control for the astrionic equipment to operate within its design limits in both natural and induced thermal environments.

The hardware defined for the thermal conditioning subsytem is as follows:

- Coolant Pump
- Service Heat Exchanger
- Coolant Accumulator
- Coolant Fluid
- Radiators
- Louvers
- Component Mounting Panels
- Multilayer Insulation
- Miscellaneous Plumbing

Each of these is described in the following sections. Figure 3-7 gives a summary of the thermal conditioning subsystem. A more detailed discussion on the selection of equipment required for the thermal conditioning subsystem is given in Appendix J to this report.

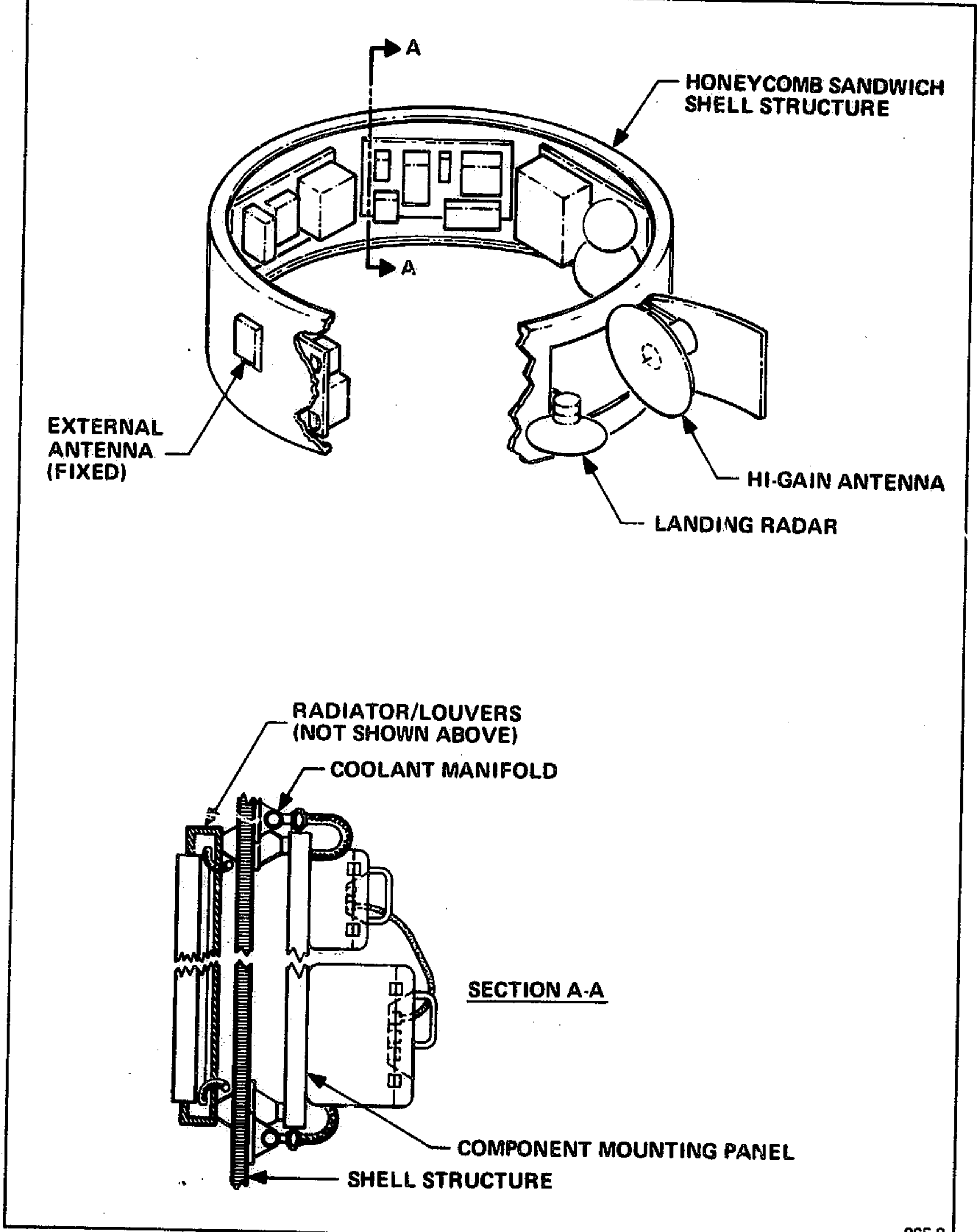

Figure 3-5. Astrionic Module Shell Structure Option

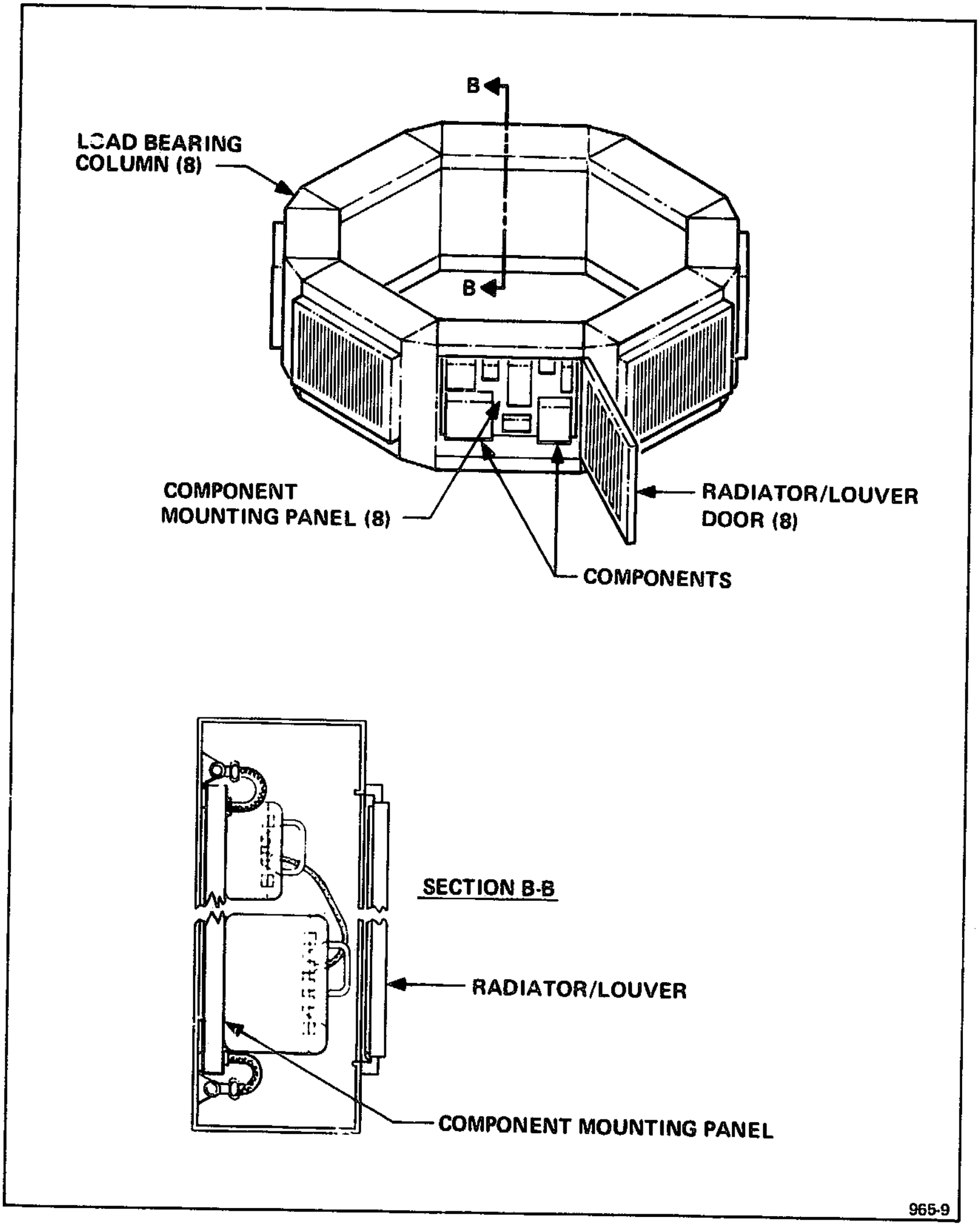

Figure 3-6. Astrionic Module Open Frame Structure Option

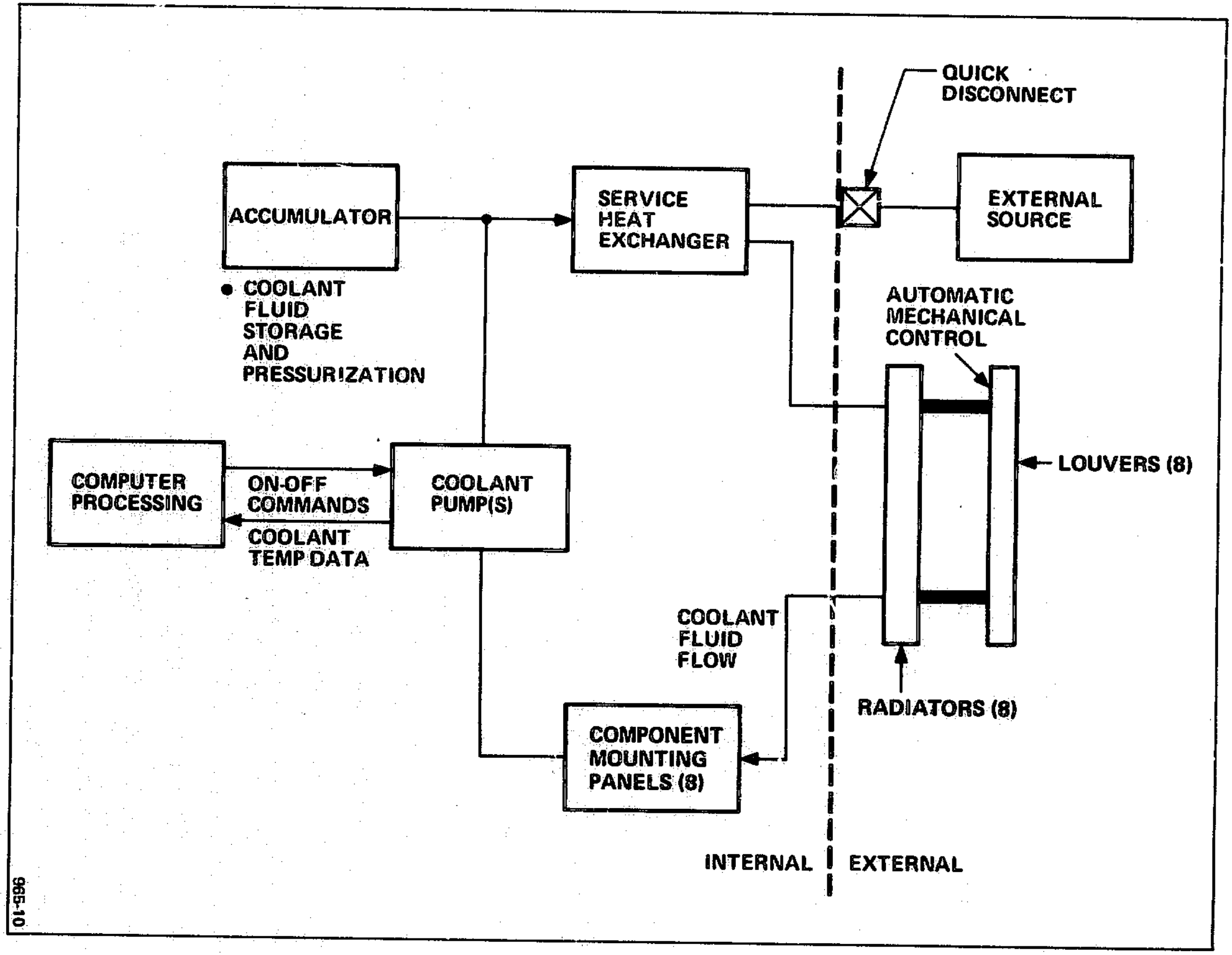

Figure 3-7. Thermal Conditioning Subsystem

3.9.1 Coolant Pump

The coolant pump is a device used to circulate the coolant fluid through a closed loop thermal control loop during the active phases of the tug mission. The circulating fluid transfers the heat from the electronic equipment to the space radiators and controls the temperature of the astrionic equipment.

3.9.2 Service Heat Exchanger

A service heat exchanger is included in this subsystem to allow thermal conditioning of the astrionic module equipment by another space element when the tug is docked with that element and for thermal conditioning by ground equipment during system checkout prior to launch into space.

3.9.3 Coolant Accumulator

The coolant accumulator provides a reservoir for the coolant fluid and the liquid pressurization for the coolant loop. The coolant accumulator selected gives mechanical pressurization using a spring loaded bellows type accumulator. The accumulator also provides volume for thermal expansion of the coolant fluid.

3.9.4 Coolant Fluid

The coolant fluid circulates through the closed loop coolant system to transfer heat from the astrionic module electronic equipment to the radiators. The coolant fluid must have a low freezing point, as well as the other desirable properties of low viscosity, high specific heat and compatibility with other materials.

3.9.5 Radiators

The space radiators reject heat directly into deep space by virtue of the temperature and optical properties of its radiating surfaces. The heat emitted by the radiators is dependent upon radiator area. As the coolant fluid passes through the radiators, heat extracted from the coolant is emitted to outer space. The radiators also aid in micrometeoroid protection for the vehicle.

Each radiator is 48 inches by 44 inches by 2 inches thick (overall envelope dimensions) and a maximum of eight radiator sections make up the space radiation system. The number of sections can vary depending upon the thermal control and micrometeoroid protection required. The space radiators are mounted to the outside of the structure by low conductivity stand-off mounts.

3.9.6 Louvers

The louvers are attached on the outside of the radiators. They are mechanical shutters which are used to integrate the active and passive cooling concepts by adjusting the optical properties of the radiating surface. The louvers also aid in micrometeoroid protection of the astrionic equipment.

Each louver section is 48 inches by 44 inches by 3 inches thick (overall envelope dimension) and a maximum of eight louver sections are used in conjunction with the space radiator sections. The number of sections can vary depending upon the thermal control and micrometeoroid protection required.

3.9.7 Component Mounting Panels

The component mounting panels are plates on which astrionic module equipment is mounted. A maximum of eight panels may be used with the number of panels determined by the area and locations required for component mounting. A component mounting panel without internal fluid passages may be used for components requiring no thermal control.

The thermal conditioning mounting panels have an inlet and outlet for the coolant fluid to flow. The fluid transfers heat from components mounted on the panel as it passes through the panel. The panels are 48 inches by 36 inches by one inch thick. The panels include mounting holes for the components.

3.9.8 Multilayer Insulation

The multilayer insulation consists of layers (the number will be determined later) of thin aluminized film. The insulation is located between the inner structure of the vehicle and the radiators. It provides the necessary thermal isolation between the astrionic module components and the external space environment required during storage phases of the tug mission.

3.9.9 Miscellaneous Plumbing

Plumbing is required for connection between the thermal control subsystem components. This will include pipes, connections, etc. to perform this task.

3.10 ONBOARD CHECKOUT SUBSYSTEM DESCRIPTION

The onboard checkout subsystem determines the operational capability of the space tug vehicle, diagnoses malfunctions and reverifies operational status after a maintenance sequence.

The checkout study has not defined any hardware to support the checkout subsystem to date. The study has determined the checkout requirements, defined the types of checkout tests to be performed and conducted a preliminary analysis of checkout methods to meet the space tug checkout requirements.

The results to date indicate that a centralized checkout approach, which uses the onboard computer for test control and data analysis, should be considered as the prime means for space tug checkout. However, built-in-test-equipment, which is test equipment built into prime equipment for malfunction isolation and checkout, offers distinct checkout advantages for unique types of hardware. Therefore, a centralized checkout approach, supplemented by built-in-test-equipment, is recommended for the checkout subsystem. A more detailed description of the checkout analysis is given in Appendix K to this report.

3.11 SUPPORTING ANALYSES

The analyses discussed in this section were used either as direct inputs to aid in selection of equipment or concepts for the astrionic subsystem or as background information for system and subsystem analysis.

3.11.1 Maintainability Considerations

The maintainability effort in this study has been to coordinate with and provide required inputs to the system and subsystem analysis and configuration. Emphasis has also been placed on test requirements, checkout function configurations, definition of the lowest replaceable units (LRU), and the ease of access for maintenance.

Results to date indicate that the replacement capability for astrionic equipment should be at two levels: the LRU level for individual components and the component mounting panel level for replacing a group of related components. Appendix L to this report discusses the test capabilities required for the maintainability philosophy and a detailed discussion of the replacement of astrionic equipment.

3.11.2 Reliability Considerations

The reliability effort has been used to factor reliability concepts into the selection of subsystem equipment for the space tug missions. Emphasis was placed on factoring redundancy into mission critical astrionic equipment.

After the astrionic system had been configured from a requirements point of view, analysis was performed to determine the relative reliability of the simplex system. Reliability enhancement was then investigated for duplex, triplex, and for the system as finally configured. The relative reliabilities were compared with an assumed reliability goal. This analysis shows that for long duration missions, especially the lunar landing mission, reliability goals cannot be achieved, even with triplexed systems. Follow-on effort must concentrate on mission groundrules concerning maintenance on the lunar surface in case of malfunction. Also, very sophisticated redundancy schemes or use of rescue missions should be addressed.

A more detailed analysis of the reliability analysis is given in Appendix M to this report.

3.11.3 Safety Assurance Considerations

The space tug safety assurance analysis identifies the parameters of the space tug vehicle which should be monitored for off-nominal conditions which could cause the loss of crew, vehicle, payload or mission and recommends action to be taken if off-nominal events occur.

The following malfunctions or conditions were identified as catastrophic for the crew, vehicle and/or payload:

- Loss of attitude control
- Guidance reference failure
- Loss of electrical power
- Reactant tanks explosion hazard
- Battery explosion hazard
- TVC actuator hardover or stationary in one position
- Loss of crew module life support system
- Loss of crew module electrical power

It has also been identified that the capability should exist to:

- Separate the crew module or payload module from the vehicle.
- Separate the crew module and astrionic module from the propulsion module.
- Use pyrotechnic devices as a backup to module interface separation.

No automatic abort requirement has been identified for any of the tug missions with the possible exception of a manned four stage Saturn V mission.

A more detailed discussion of the space tug safety analysis is given in Appendix N to this report.

3.11.4 Display Considerations

The display function will provide the displays and control required by the crew on manned missions to perform their assigned mission. The effort to date has assumed two flight stations and a housekeeping or experiment station. The switches, lights and dials used on present Apollo missions were rejected for the display function and effort has concentrated on multipurpose electronic displays to meet the display requirements. Three cathode ray tubes, along with associated equipment, was selected for the display function. A more detailed discussion of the analysis to date is included in Appendix O of this report.

3.11.5 Radiation Effects Impact

Radiation effects on the space tug astrionics have been assessed relative to natural and induced radiation environments, their effects on components, circuits and systems, radiation hardening techniques, a systems hardening plan and penalties associated with the radiation hardening.

For space tug missions now being considered, natural radiation environments have little impact on the design of space tug astrionics. However, in the case of energetic solar flare activity, space tug missions may have to be delayed or terminated early, thus imposing an operational constraint.

Interaction of the space tug with induced nuclear reactor radiation from NERVA and space base power reactors will impose minimum impact on tug astrionic design. Here again, however, tug operational safety constraints will have to be met in order to keep the tug in zones of low radiation environments.

By far, the most stringent environment that the tug could be exposed to is that of pulsed radiation from sources external to the tug. Techniques exist to harden electronics to this type environment, but will impact cost, weight, power requirements, etc. These hardening techniques must be considered in the initial design of the space tug astrionic module. To harden a system once the system approach and technologies to implement this approach have been finalized becomes a formidable task.

A more detailed discussion of the effects of radiation on the space tug astrionics is given in Appendix P to this report.

3.11.6 Cost Impact

A rough order of magnitude (ROM) costing exercise was performed for the space tug astrionic module for selected missions. A work breakdown structure (WBS) was developed for the space tug to aid in the tiering of costs for the astrionic module. The astrionic module components, such as the fuel cell, were costed and summed into subsystem cost, astrionics cost and finally astrionic module costs.

Unit recurring cost for a representative Reusable Synchronous Mission Tug Astrionic Module is $8 million which may be prorated over ten missions to $1.52 million per mission. Nonrecurring costs are estimated at $87 million which includes $27 million for two flight test articles.

Appendix Q of this report gives a detailed analysis of the costing exercise and the resulting ROM cost impacts.

3.11.7 Orbital Lifetime Consideration

A study was performed to determine how long the tug could be stored in a 100 N.M. circular parking orbit in a fueled and unfueled state. To obtain the results, a trajectory integration program, which contains an atmospheric drag model, was utilized. Lifetime was determined by integrating the initial vehicle state in a 100 N.M. circular orbit until the perigee radius of the orbit decreased below 50 km. It was assumed that the tug's angular attitude was not controlled during the altitude decay and that tumbling would occur. Net lift forces were zero. This means that the tug would be tumbling, and thus there would be no net lift forces acting on it. The ballistic coefficient, β, for the tug was considered to be similar to that of the S-II stage of the Saturn V vehicle.

$$\beta = \frac{C_D A}{2m}$$

where: C_D = drag coefficient

A = area of vehicle projected onto a plane normal to the relative velocity vector

m = mass of vehicle

The orbital lifetime as a function of the ballistic coefficient is shown in Figure 3-8. The lifetimes of the tug in empty and full configurations are indicated on the plot and tabulated below.

Configuration	Lifetime
EMPTY:	18-20 hrs
FULL:	100-120 hrs

An estimate was made as to how much total velocity impulse would be necessary to maintain or station keep the tug in the 100 N.M. circular parking orbit. The estimate was made by assuming that the amount of orbital energy lost per revolution would be replaced by adding a velocity impulse to offset the loss. The results are presented below.

Configuration	Impulse per Revolution
EMPTY:	150 m/sec
FULL:	30 m/sec

Additional fuel would also be necessary for attitude control.

In summary the following statements can be made about lifetime considerations at a 100 N.M. parking orbit.

- Lifetimes are relatively short compared with requirements for the duration of quiescent tug storage.
- Stationkeeping fuel requirements at 100 N.M. are excessively high.

4.0 SYSTEM DESCRIPTION

4.1 SYSTEM FUNCTIONAL DESCRIPTION

The astrionic subsystem descriptions were given in Section 3.0. These subsystems are connected and controlled by a centralized computing capability and a data bus with its associated standard interface units (SIU). A layout of the total astrionic system is shown in Figure 4-1.

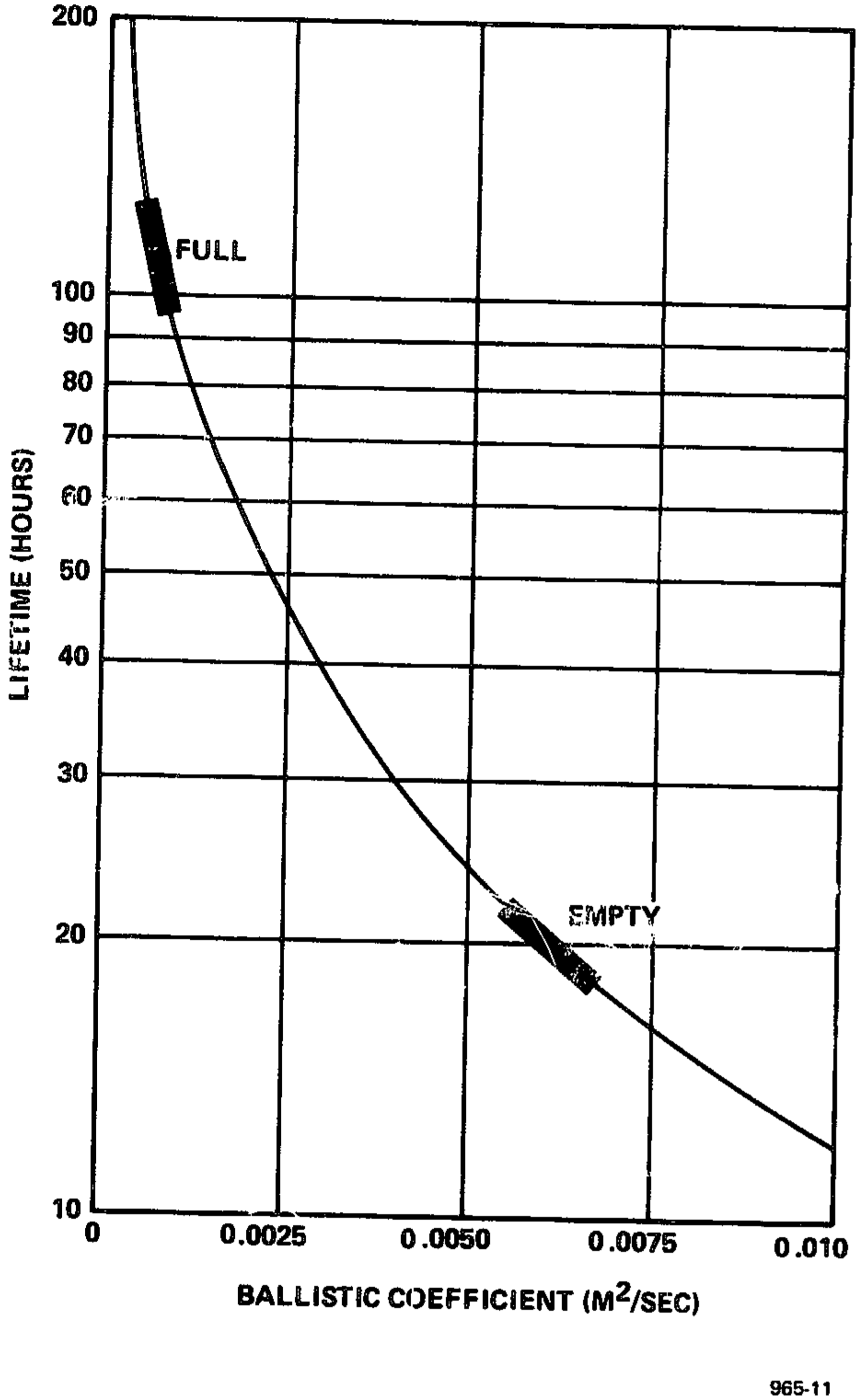

Figure 3-8. Space Tug Orbital Lifetime

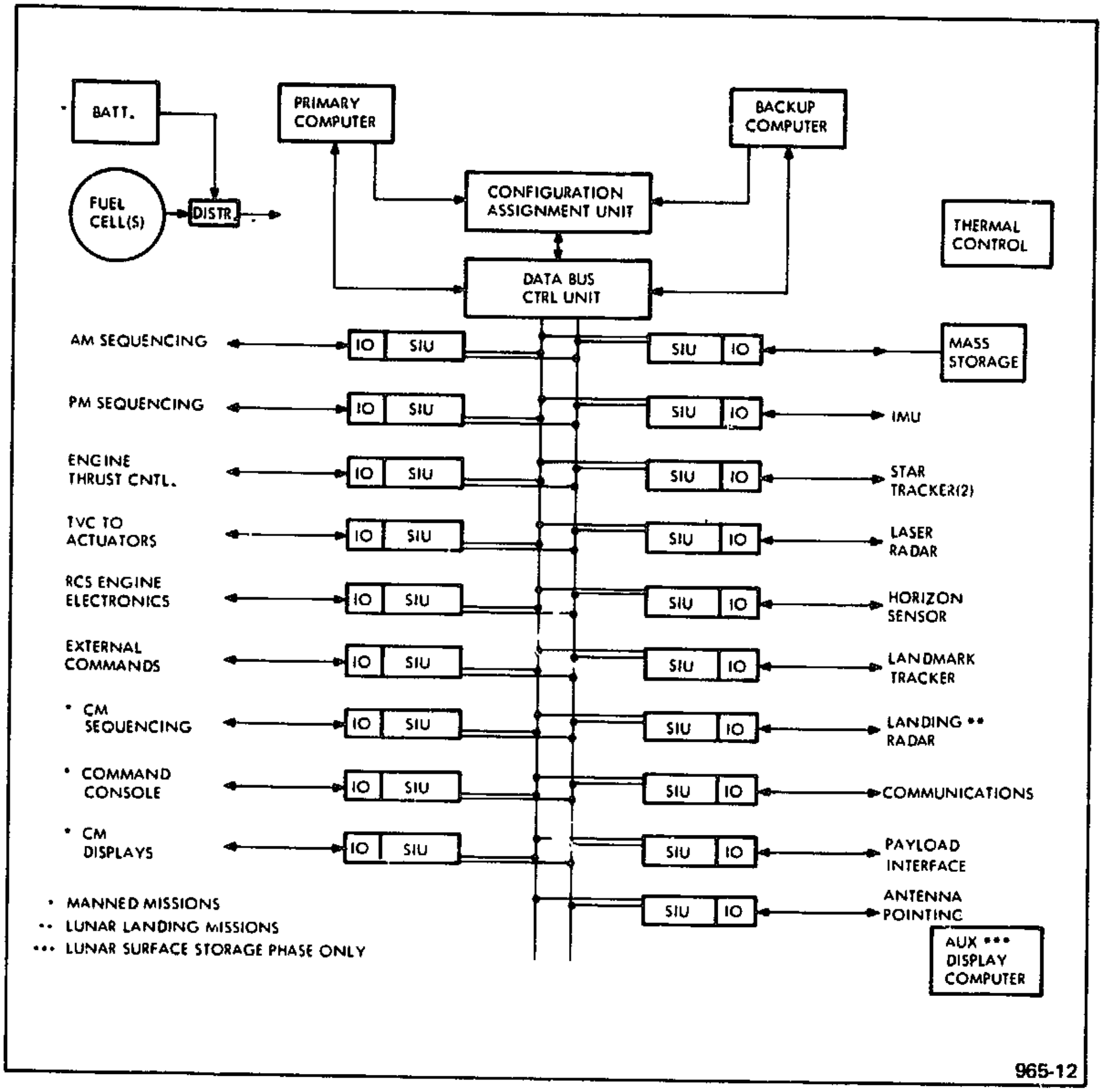

Figure 4-1. Space Tug Astrionic System Layout

The system is configured for a centralized, rather than a federalized, approach to data management. The data bus control unit controls the serial, digital data on the data bus and is the interface between the data bus and the computers. The subsystem equipment is connected to the data bus by standard interface units. These SIUs are designed to permit unlimited exchange of equipment on the data bus so that no electrical timing or loading perturbations are introduced on the rest of the system. The Input/Output (I/O) processors are used to adapt the subsystem equipment to the SIUs and the data bus in turn. In new designs, the I/O can be an integral part of the equipment. This approach minimizes impact of the introduction of new state-of-the-art hardware as the astrionic system matures. Also shown on this figure is the fuel cell used for supplying electrical power to the astrionic equipment and the thermal control subsystem for thermal control of the components.

The subsystem equipment on the data bus supplies data to and receives data from the data management processing capability. Figures 4-2 through 4-7 show block diagrams and functional interconnections for the subsystems. No attempt was made to show equipment power connections, details of checkout, telemetry signals or redundant information flow in these figures.

Figure 4-2 shows the functional interconnection diagram for the navigation sensors. This figure shows how the sensors required for the synchronous orbit mission interface with their associated electronic units and the standard interface units. The signals generated by the navigation sensors are fed to and from the computer for processing of navigation data. The data received from the navigation sensors is processed by the computer for use by other space tug functional areas, especially guidance and control.

The guidance and control functional interconnection diagram is shown in Figure 4-3. As is shown, inputs from the navigation function are processed in the computer in conjunction with guidance scheme computations. After the guidance and navigation computations, the results are used for guidance function processing in the computer. The control commands are sent through the data bus and SIUs to the RCS subsystem and the main propulsion engine. Feedback loops provide additional inputs back to the computer for processing, enabling the guidance and control function to steer the vehicle.

A functional block diagram of the command and control subsystem is shown in Figure 4-4. The functional interconnects are shown for the components in this subsystem. The telemetry data is processed and formatted by the data management subsystem. This data is fed through the data bus and SIU for modulation processing prior to transmission. Command data is received by the command and control subsystem, demodulated, decoded and sent through the data bus to the computer processing. The control signals for the command and control equipment shown on this diagram are also received from the data bus.

The distribution and functional interconnects for the electrical power subsystem are shown in Figures 4-5 and 4-6. Figure 4-5 shows how electrical power from the fuel cells is connected to other astrionic components. Figure 4-6 shows in more detail the "A" bus distribution and interconnections. Also shown are the fuel cell interfaces with the data management computer and signals to the propulsion module.

Figure 4-7 shows the functional interconnects for the thermal control subsystem. As shown, the computer supplies signals to operate the coolant pumps and receives temperature and pressure signals required for proper operation.

When the functional interconnects are integrated into an astrionic system by the data bus, SIUs and data management subsystem, proper and efficient operation at the system level is maintained.

4.2 EQUIPMENT USAGE FOR SPACE TUG MISSIONS

The astrionic system components used for the space tug missions are listed in Table 4-1. This table shows the number or quantity of components or hardware required for every vehicle of each design mission used in this study. The number of components are classified as primary (P) or backup (B). The primary units are to be operated when that particular piece of equipment is required during the mission. The backup units are a backup to the primary unit and are used when the primary unit has a failure. The backup units may be either inactive or in a standby mode depending on the mission phase and the criticality of the equipment.

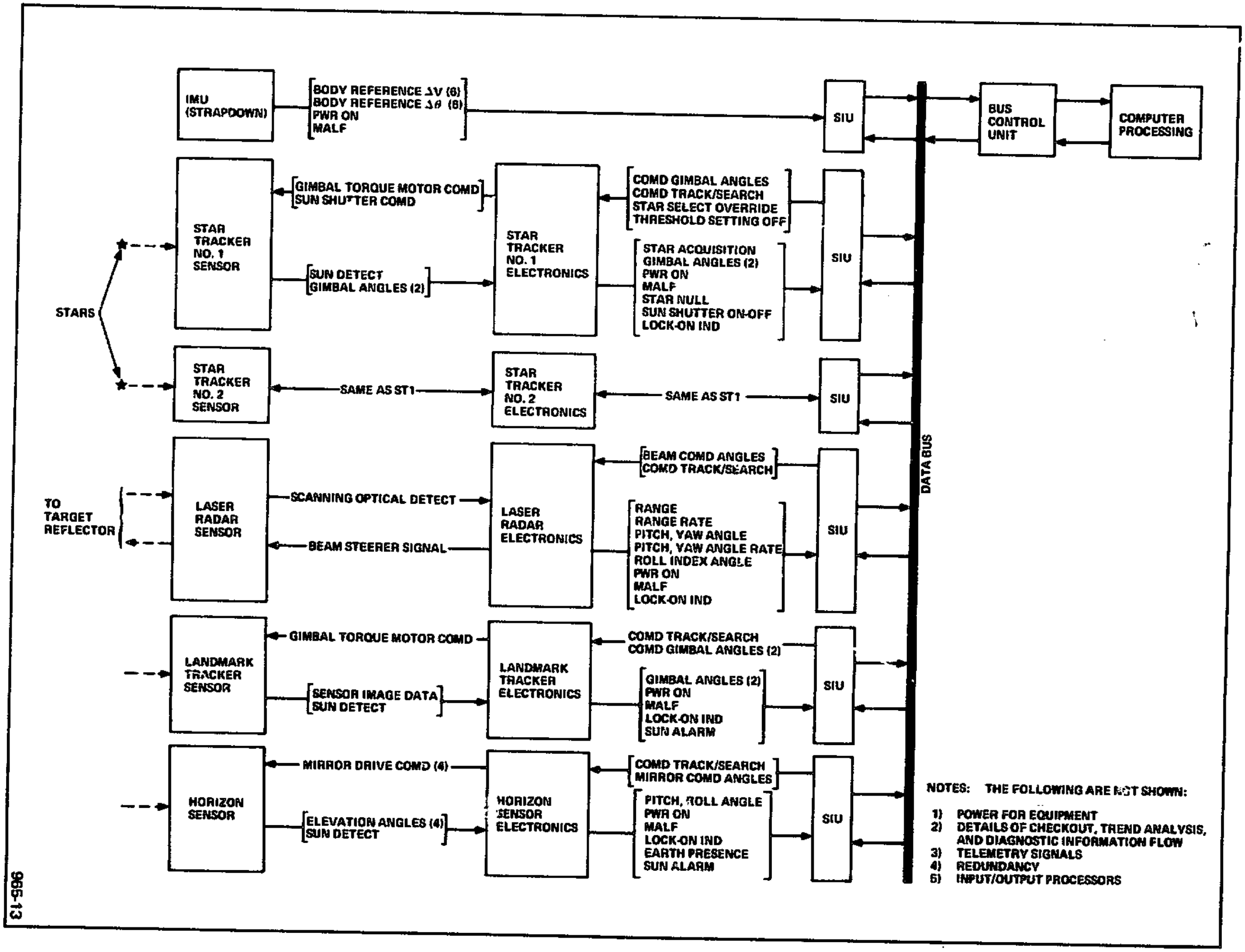

Figure 4-2. Navigation Functional Interconnection Diagram (Synchronous Orbit Mission)

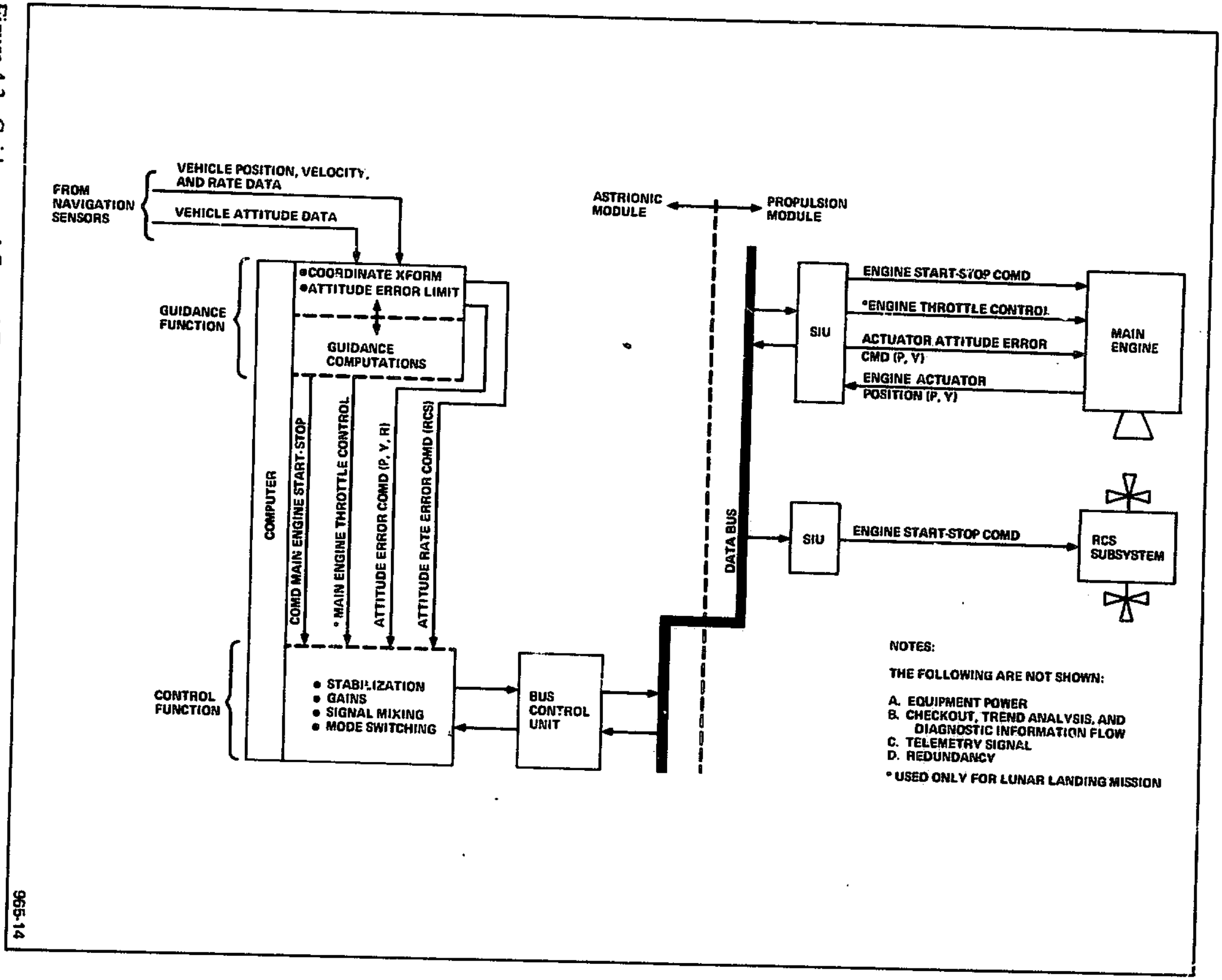

Figure 4-3. Guidance and Control Functional Interconnection Diagram

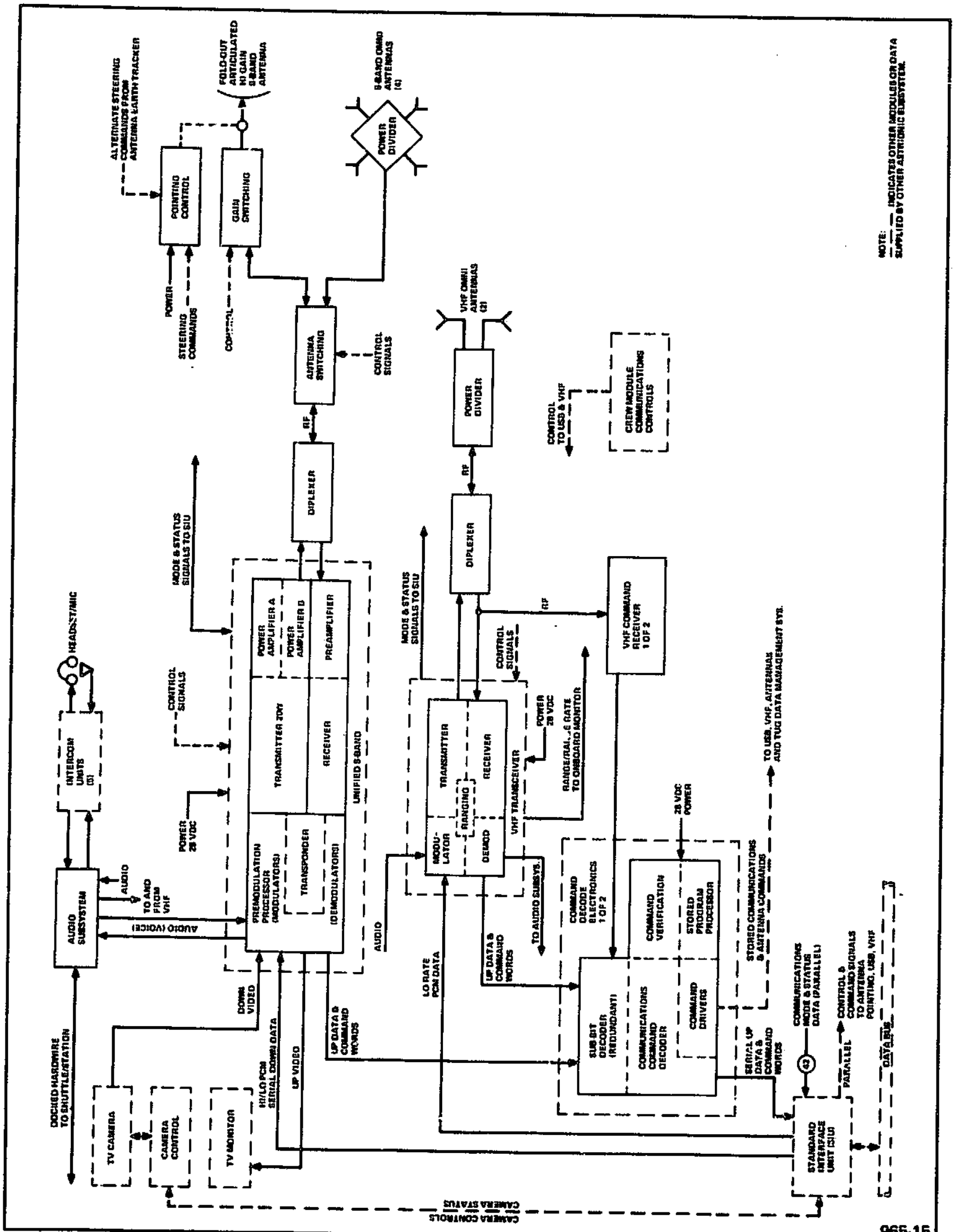

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Figure 4-4. Composite Tug Command and Control Subsystem Preliminary Functional Block Diagram

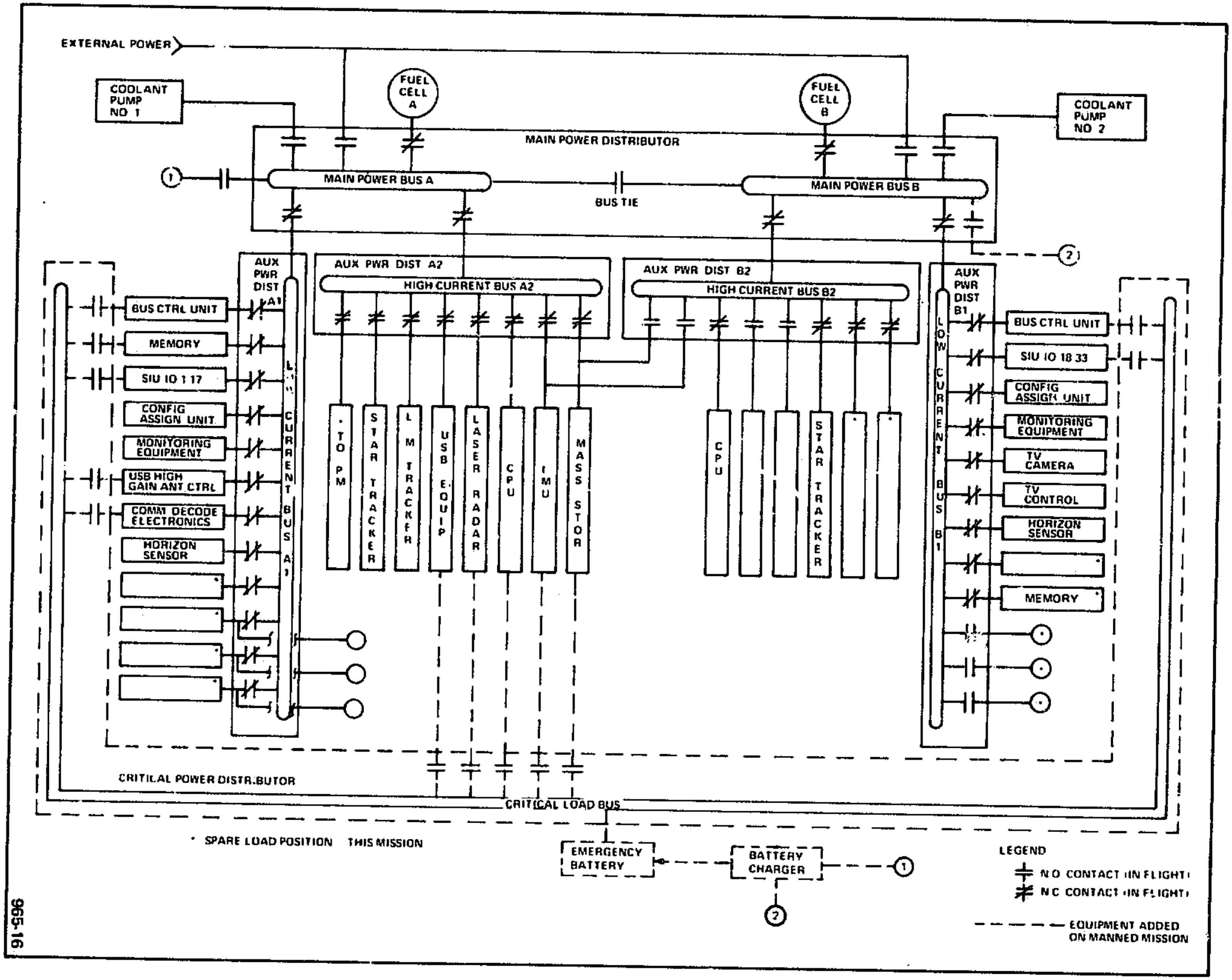

Figure 4-5. Electrical Networks Subsystem (Power Distribution)

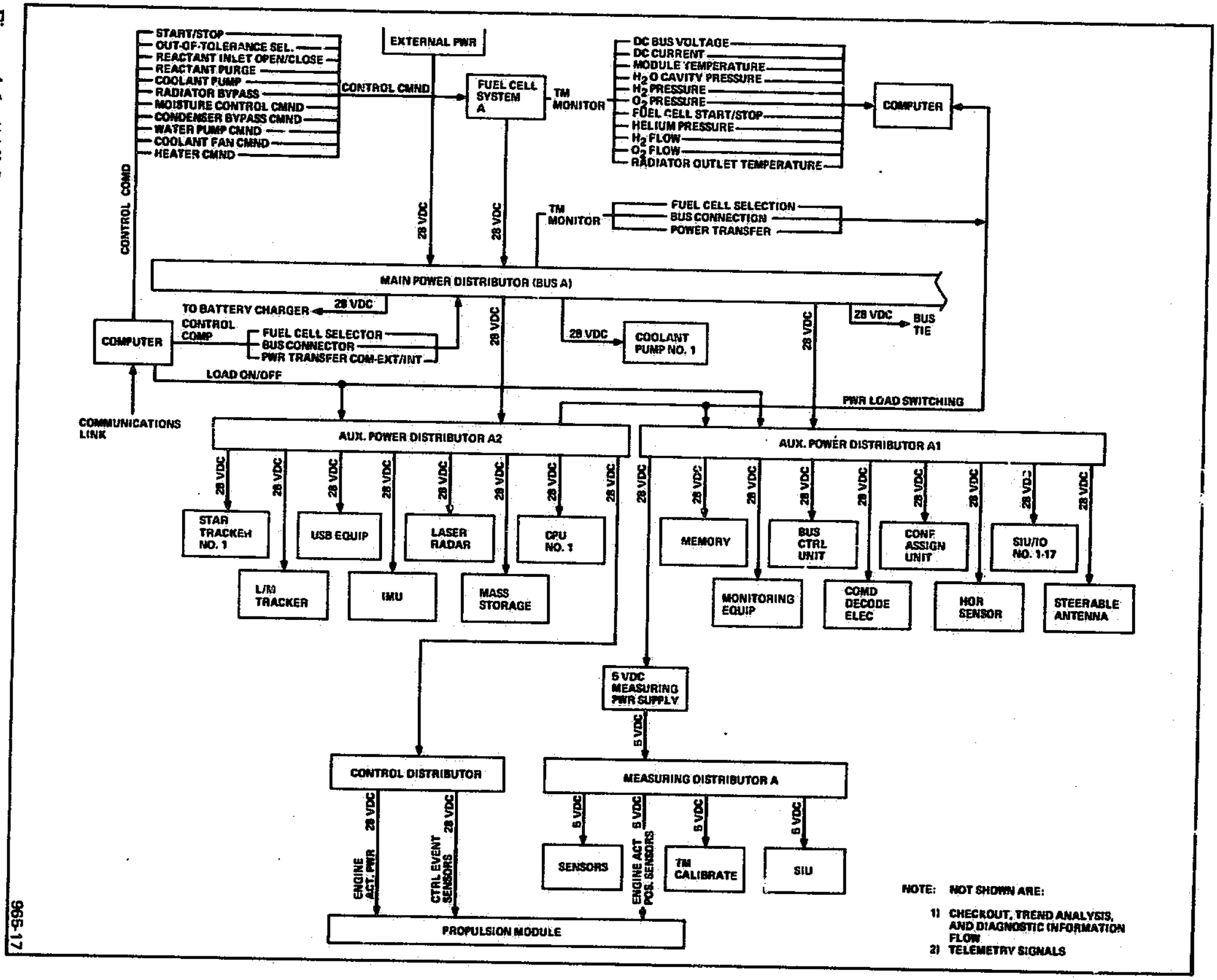

Figure 4-6. "A" Side EPS Functional Interconnect Space Tug

Table 4-1. Space Tug Equipment Usage (Sheet 1 of 2)

COMPONENT	MISSION																			
	SYNCHRONOUS (EXPENDABLE)		SYNCHRONOUS (REUSABLE – 1ST TUG)		SYNCHRONOUS (REUSABLE – 2ND TUG)		LUNAR LANDING		EARTH ORBITAL OPERATION		LUNAR ORBITAL OPERATION		PLANETARY (REUSABLE – 1ST TUG)		PLANETARY (EXPENDABLE – 2ND TUG)		REUSABLE NUCLEAR SHUTTLE		FOUR STAGE SATURN V	
	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B
CPU	1		1	1	1	1	1	2	1	2	1	2	1	1	1	1	1	1	1	1
BUS CTRL UNIT	1		1	1	1	1	1	2	1	2	1	2	1	1	1	1	1	1	1	1
MAIN MEMORY	1		2		2		2		2		2		2		2		2		2	
MAG. TAPE UNIT			1		1		1		1		1		1		1		1		1	
DISPLAY MEMORY							1		1		1									
C. ASSIGN. UNIT			1	1	1	1	1	2	1	2	1	2	1	1	1	1	1	1	1	1
AUX. MONT. COMPUTER							1													
IMU	1		1		1		1		1		1		1		1		1		1	
LASER RADAR	1		1		1		1		1		1		1				1		1	
STAR TRACKER	2		2		2		2		2		2		2		2		2		2	
LANDMARK TRACKER	1		1		1		1		1		1		1				1		1	
HORIZON SENSOR	1				1								1				1		1	
LANDING RADAR							1													
RATE GYROS																			1	
ACCELEROMETERS																			1	
FUEL CELL (2KW)	1		2		2		2		2		2		2		1		2		2	
REACTANT – H_2(LBS)	3		2		11		5		4		4		4		1		54		17	
REACTANT – O_2(LBS)	17		11		89		43		36		36		36		8		486		161	
H_2 TANK	1		2		2		2		2		2		2		1		2		2	
O_2 TANK	1		2		2		2		2		2		2		1		2		2	
BATTERY								1		1		1					2			
DC REGULATOR	1		2		2		2		2		2		2		1		2		2	
BATTERY CHARGER							1		1		1						1			
SIU	19		27	30	25	28	36	39	35	38	35	38	29	32	28	31	29	32	29	32
DATA BUS	1		1	2	1	2	1	2	1	2	1	2	1	2	1	2	1	2	1	2
MONITORING UNIT	12		12	12	12	12	18	18	18	18	18	18	12	12	12	12	12	12	12	12
AUX. MONT. UNIT							3													
POWER DISTR	1		1		1		1		1		1		1		1		1		1	
AUX. POWER DISTR	4		4		4		4	1	4	1	4	1	4		4		4	1	4	
WIRE & CABLES (LBS)	150		180		180		200		200		200		180		150		200		180	
JUNCTION BOXES	8		8		8		8		8		8		8		8		8		8	

Table 4-1. Space Tug Equipment Usage (Sheet 2 of 2)

COMPONENT	MISSION: SYNCHRONOUS (EXPENDABLE)		SYNCHRONOUS (REUSABLE – 1ST TUG)		SYNCHRONOUS (REUSABLE – 2ND TUG)		LUNAR LANDING		EARTH ORBITAL OPERATION		LUNAR ORBITAL OPERATION		PLANETARY (REUSABLE – 1ST TUG)		PLANETARY (EXPENDABLE – 2ND TUG)		REUSABLE NUCLEAR SHUTTLE		FOUR STAGE SATURN V	
	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B	P	B
USB ELECT.	1		1		1		1		1		1		1		1		1		1	
USB DIPLEXER	1		1		1		1		1		1		1		1		1		1	
USB ANTENNA SWITCH	1		1		1		1		1		1		1		1		1		1	
USB POWER DIVIDER	1		1		1		1		1		1		1		1		1		1	
USB OMNI ANTENNAS	4		4		4		4		4		4		4		4		4		4	
USB HI-GAIN ANT. CTRL	1		1		1		1		1		1		1		1		1		1	
USB HI-GAIN ANT	1		1		1		1		1		1		1		1		1		1	
VHF TRANSCEIVER							1		1		1									
VHF DIPLEXER			1		1		1		1		1									
VHF POWER DIVIDER			1		1		1		1		1									
VHF OMNI ANT.			2		2		2		2		2									
CMD. DECODE ELEC.	1		2		2		1	1	2		1	1	1	1	1	1	1	1	1	1
TV CAMERA	1		1		1		1		1		1		1		1		1		1	
TV CAMERA CTRL	1		1		1		1		1		1		1		1		1		1	
VHF COMMAND RCVR			2		2				2											
AUDIO EQUIPMENT							1		1		1									
COOLANT PUMP	1	1	1	1	1	1	TO BE DETERMINED		1	1	1	1	1	1	1	1	1	1	1	1
SERVICE HEAT EXCHANGER	1		1		1				1		1		1		1		1		1	
COOLANT ACCUM.	1		1		1				1		1		1		1		1		1	
RADIATOR	6		8		8				8		8		8		6		8		8	
COOLANT FLUID (LBS)	50		50		50				50		50		50		50		50		50	
LOUVERS	6		8		8				8		8		8		6		8		8	
COMP. MTG PANEL	6		8		8				8		8		8		6		8		8	
MISC. PLUMBING (LBS)	60		60		60				60		60		60		60		60		60	
MULTILAYER INSUL. (LBS)	25		25		25				25		25		25		25		25		25	
MOUNTING HARDWARE (LBS)	20		20		20		20		20		20		20		20		20		20	
STRUCTURE (LBS)	400		400		400		400		400		400		400		400		400		400	

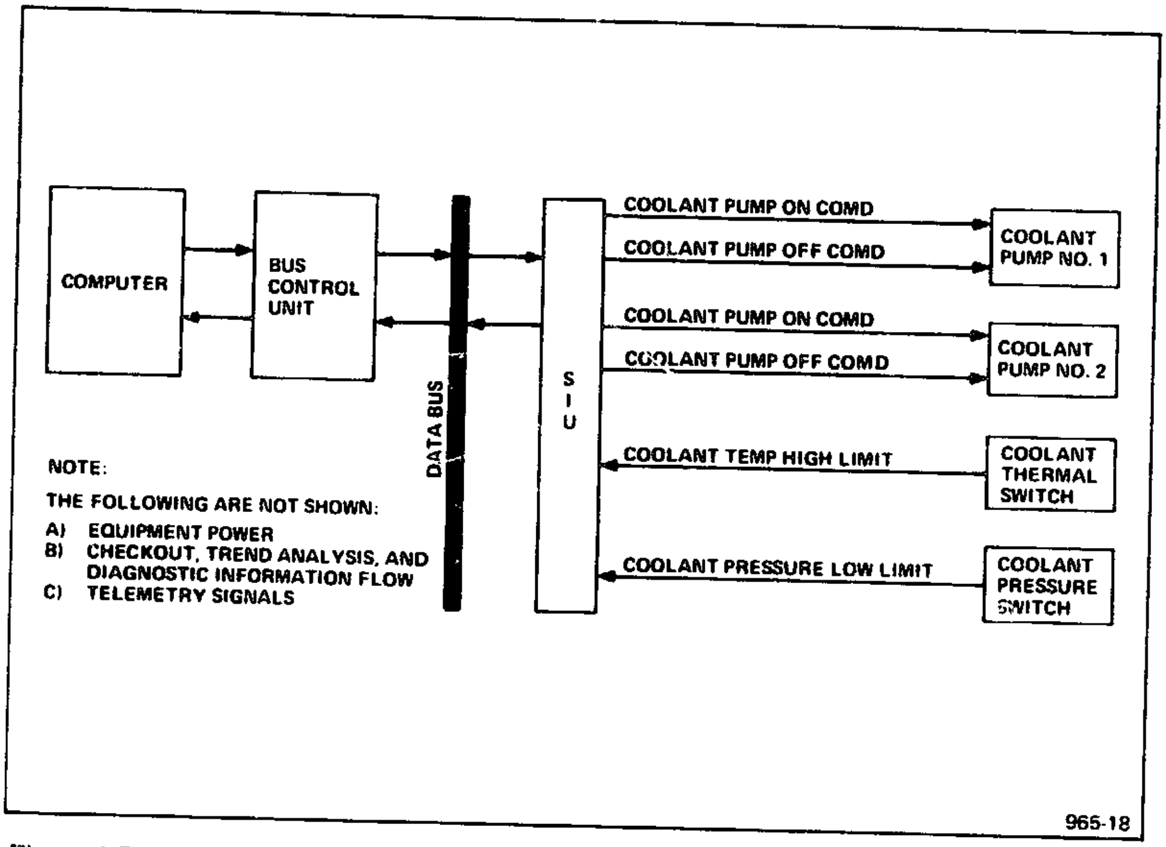

Figure 4-7. Astrionic Module Thermal Control Functional Interconnection Diagram

4.3 PHYSICAL CHARACTERISTICS OF ASTRIONIC EQUIPMENT

Tables 4-2 through 4-7 define the physical characteristics of each of the hardware items defined during this study. These tables detail the dimensions in inches of the items in width, height and depth, the weight in pounds of each of the components, the active and standby power requirements in dc watts and the operating temperature limits (where available) in Fahrenheit degrees.

Some of the physical characteristics given in the figures are based on actual built hardware; others are based on typical characteristics for that type of component. The physical characteristics were used in equipment layouts for the space tug astrionic module to show the feasibility of the astrionic module concept (See Figures 4-8 and 4-9). The item numbers shown on the physical characteristics sheets refer to the equipment shown in Figure 4-8. As these figures show, sufficient space is available for equipment used for the worst case mission (lunar landing) as far as equipment is concerned. Future additional studies or reliability considerations may increase the components required in the astrionic module. But the capability exists for growth capability in the module and some increase can be handled.

Table 4-2. Space Tug Astrionic Module Equipment Characteristics–Data Management Subsystem

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES)			WT. (LBS)	POWER (WATTS-DC)		OPERATING TEMP. LIMITS (°F)
		W	H	D		(A)	(S)	
10	CPU	13.4	9.4	7.0	60.0	80	80	-40° TO 131°
19	BUS CONTROL UNIT	12.0	8.0	3.6	35.0	50	50	-40° TO 131°
20	MAIN MEMORY	10.0	6.0	5.0	20.0	100	10	-40° TO 131°
21	MAGNETIC TAPE UNIT	10.0	7.0	5.0	10.0	15	0	32° TO 130°
57	DISPLAY MEMORY	5.0	6.0	5.0	10.0	50	5	-40° TO 131°
38	CONFIG. ASSIGNMENT UNIT	8.0	6.0	3.0	12.0	15	15	-40° TO 131°

A = ACTIVE
S = STANDBY

Table 4-3. Space Tug Astrionic Module Equipment Characteristics–Navigation, Guidance and Control Subsystem

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES)			WT. (LBS)	POWER (WATTS-DC)		OPERATING TEMP. LIMITS (°F)
		W	H	D		(A)	(S)	
9	IMU	13φ	9.0		80	200		0° TO 50°
	LASER RADAR				28.0	30		0° TO 50°
4	• SENSOR	8φ	17.0		20.0			0° TO 50°
14	• ELECTRONICS	8.0	12.0	12.0	8.0			0° TO 50°
	STAR TRACKER				23.0	20 (30P)		0° TO 50°
1	• SENSOR	9.1	8.1	6.4	14.0			0° TO 50°
11	• ELECTRONICS	5.5	8.1	4.0	9.0			0° TO 50°
	LANDMARK TRACKER				30.0	40		0° TO 50°
2	• SENSOR	4.0	4.0	12.0	22.0	25		0° TO 50°
12	• ELECTRONICS	8.0	8.0	12.0	8.0	15		0° TO 50°
	HORIZON SENSOR				17.0	10 (13P)		0° TO 50°
5	• SENSOR(2)	5.5	4.0	7.5	6.0			0° TO 50°
15	• ELECTRONICS	6.0	2.9	6.5	5.0			0° TO 50°
	LANDING RADAR				34.0	80		0° TO 50°
3	• SENSOR & GIMBALS	21φ	13.0		15.0			0° TO 50°
13	• ELECTRONICS				19.0			0° TO 50°
	RATE GYROS	7.0	3.6	7.0	11.0	45		-40° TO 74°C
	ACCELEROMETERS	4.0	3.6	5.0	4.0	8.4		-73° TO 71°C

A = ACTIVE
S = STANDBY
P = PEAK POWER

Table 4-4. Space Tug Astrionic Module Equipment Characteristics–Electrical Power Subsystem

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES) W	DIMENSIONS (INCHES) H	DIMENSIONS (INCHES) D	WT. (LBS)	POWER (WATTS-DC) (A)	POWER (WATTS-DC) (S)	OPERATING TEMP. LIMITS (°F)
32	FUEL CELL	16.0	21.0	32.0	169	100*		160° TO 220° OPERATING 35° TO 220° STORAGE
35	REACTANT – H_2							
36	REACTANT – O_2							
33	H_2 TANK	17.0	17.0	18.0	7.0			
34	O_2 TANK	14.0	14.0	15.0	14.0			
31	BATTERY	8.8	17.5	10.0	140.0			
43	DC REGULATOR	8.0	6.0	4.0	5.0			40° TO 90°
37	BATTERY CHARGER	6.0	6.0	3.0	8.0			

A = ACTIVE
S = STANDBY
*FUEL CELL PARASITIC POWER

Table 4-5. Space Tug Astrionic Module Equipment Characteristics Electrical Networks Subsystem

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES) W	DIMENSIONS (INCHES) H	DIMENSIONS (INCHES) D	WT. (LBS)	POWER (WATTS-DC) (A)	POWER (WATTS-DC) (S)	OPERATING TEMP. LIMITS (°F)
23	SIU	4.3	3.0	2.8	0.4	1	1	-67° TO 257°
24	DATA BUS		173 CM^3		7.5			-40° TO 131°
22	MONITORING UNIT	4.0	2.0	2.0	0.5	2	2	-40° TO 131°
22	AUX. MONT. UNIT	4.0	2.0	2.0	0.5	2	2	-40° TO 131°
30	POWER DISTRIBUTOR	12.0	17.0	8.0	39.0			
39	AUX. POWER DISTR.	11.0	9.0	6.0	13.0			
29	WIRE & CABLES (LBS)				150 TO 200			
16	JUNCTION BOXES	6.0	6.0	2.0	2.0			

A = ACTIVE
S = STANDBY

The equipment described in this section is mounted to component mounting panels. These panels also serve as "cold plates" for thermal control. The panels are in turn connected to the interior of the astrionic module structure (See Figure 4-10) for the astrionic module configuration. On the outside of the honeycomb structure are mounted the radiators and louvers for thermal control. There are eight component mounting panels and eight radiator/louver panels. This type of physical configuration provides the basis for mounting the astrionic components.

Table 4-6. Space Tug Astrionic Module Equipment Characteristics–Command and Control Subsystem

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES)			WT. (LBS)	POWER (WATTS-DC)		OPERATING TEMP. LIMITS (°F)
		W	H	D		(A)	(S)	
44	USB ELECTRONICS	21.0	7.0	8.0	42.0	107		
45	USB DIPLEXER	4.0	3.0	1.0	0.5	–		-4° TO 167°
46	USB ANT. SWITCH	2.0	1.0	2.0	0.3	–		-4° TO 167°
47	USB POWER DIVIDER	3.0	1.0	2.0	0.5			-4° TO 167°
6	USB OMNI ANTENNAS	6.0	3.0	3.0	0.5			-4° TO 167°
18	USB HI-GAIN ANT. CTRL.	10.0	8.0	5.0	10.0	10		-4° TO 167°
8	USB HI-GAIN ANT.	48.0	48.0	12.0	20.0			-4° TO 167°
27	VHF TRANSCEIVER	15.0	7.0	5.0	15.0	25		-4° TO 167°
48	VHF DIPLEXER	4.0	3.0	4.0	1.0			-4° TO 167°
49	VHF POWER DIVIDER	3.0	1.0	2.0	0.5			-4° TO 167°
7	VHF OMNI ANTENNA	2.0	11.0	5.0	1.2			-4° TO 167°
[illegible]0	COMMAND DECODE ELEC.	8.0	5.0	4.0	10.0	5	3	-4° TO 167°
41	TV CAMERA	2.0	6.0	10.0	5.0	6		-4° TO 167°
42	TV CAMERA CONTROL	3.0	3.0	2.0	2.0	3		-4° TO 167°
40	VHF COMMAND RCVR.	4.0	6.0	2.5	2.0	1	.5	-4° TO 167°
17	AUDIO EQUIPMENT	4.0	2.0	1.0	2.0	3		-4° TO 167°

A = ACTIVE
S = STANDBY

Table 4-7. Space Tug Astrionic Module Equipment Characteristics–Thermal Control Subsystem and Structures

ITEM NO.	NAME OF COMPONENT	DIMENSIONS (INCHES)			WT. (LBS)	POWER (WATTS-DC)		OPERATING TEMP. LIMITS (°F)
		W	H	D		(A)	(S)	
52	COOLANT PUMP	6.0	8.0	6.0	25.0	150		
53	SERVICE HEAT EXCH.	6.0	5.0	3.0	10.0			
54	COOLANT ACCUMULATOR	10.0	10.0	12.0	[illegible]			
55	RADIATOR	48.0	44.0	2.0	18.4			
58	COOLANT FLUID				50.0			
61	LOUVERS	48.0	44.0	3.0	11.0			
62	COMPONENT MTG PANEL	36.0	48.0	1.5	25.0			
63	MISC. PLUMBING				60.0			
	MULTILAYER INSULATION				25.0			
	MOUNTING HARDWARE				20.0			
	STRUCTURES				400.0			

A = ACTIVE
S = STANDBY

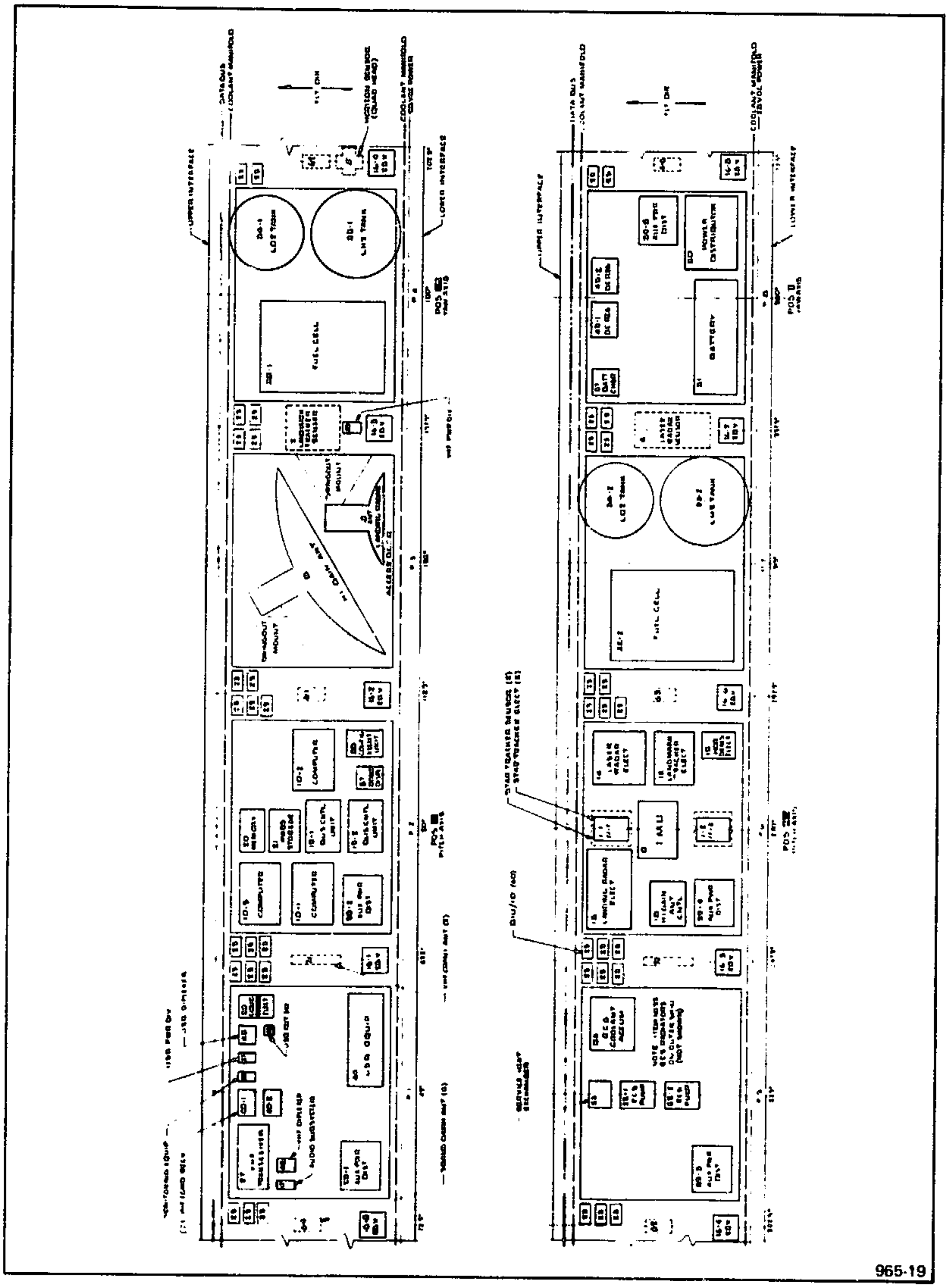

Figure 4-8. Astrionic Module Component Layout

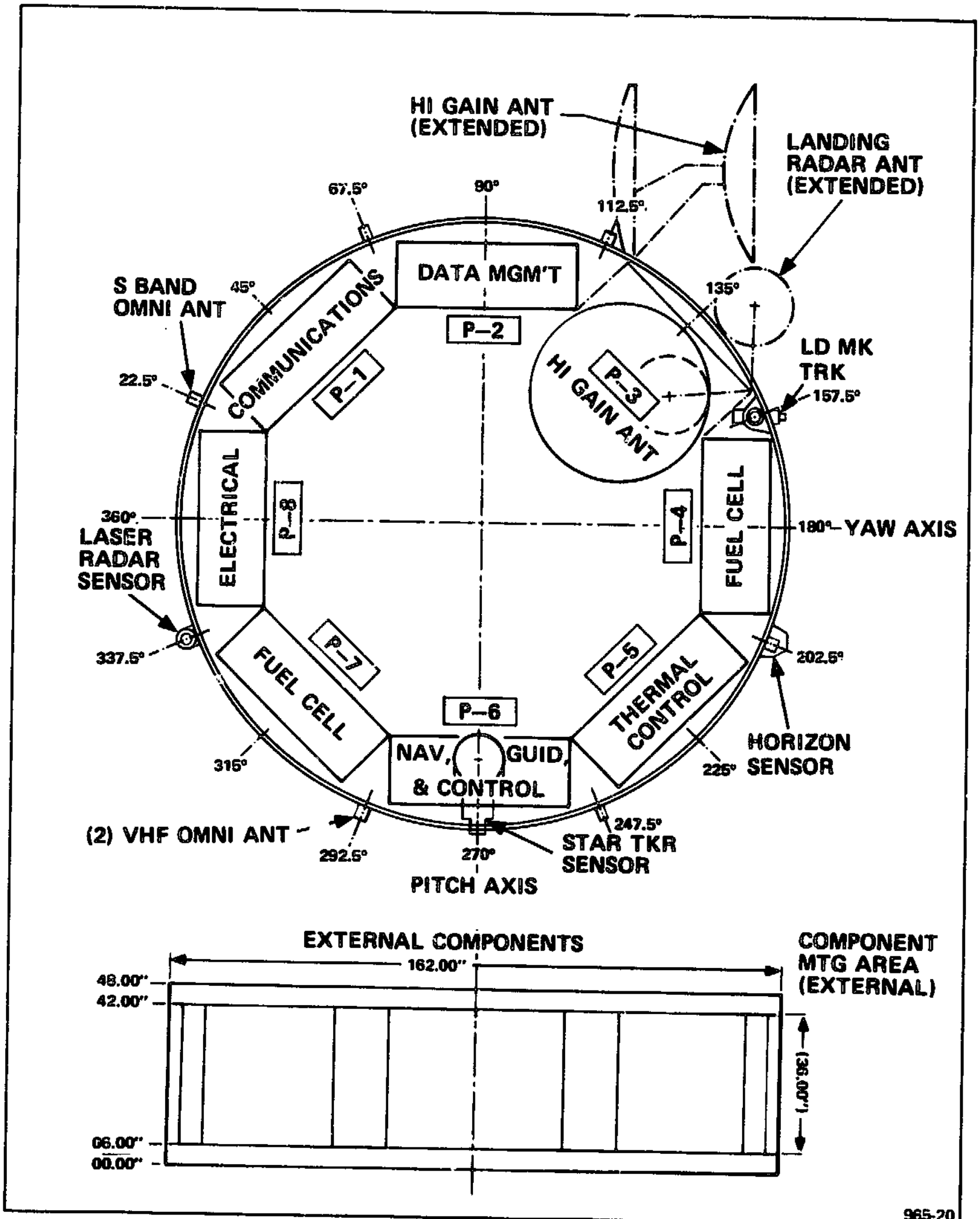

Figure 4-9. Astrionic Module Component Layout (Top View)

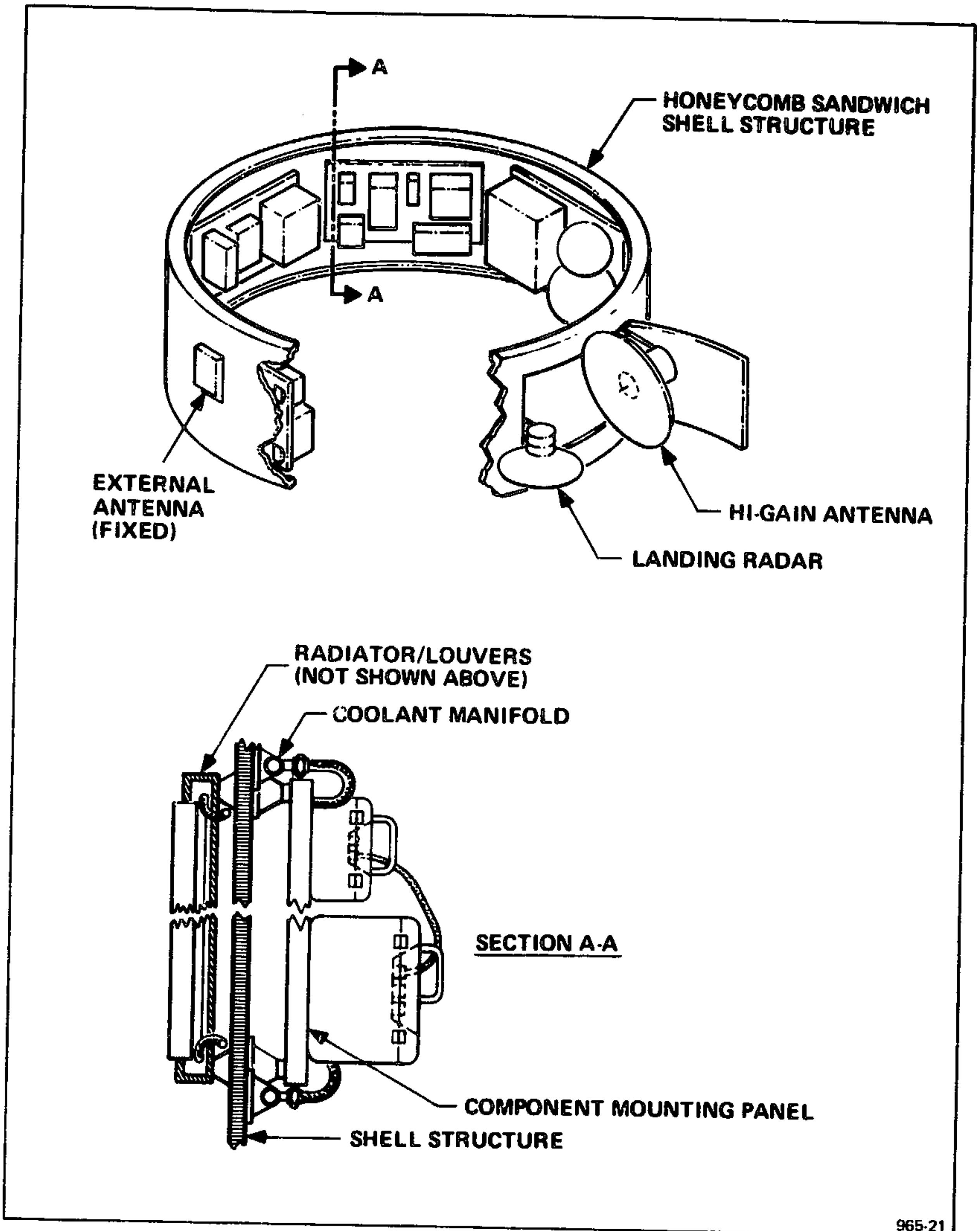

Figure 4-10. Astrionic Module Shell Structure

4.4 ASTRIONIC MODULE WEIGHT AND POWER SUMMARY

The equipment usage shown in Table 4-1 and the equipment physical characteristics shown in Tables 4-2 through 4-7 were used to develop the astrionic module weight and power summaries for the space tug missions. All units used in the astrionic module were considered for the weight summaries. The active power requirements for all primary units of the astrionic module were summarized to show the power requirement. Although all primary units may never be operating simultaneously and backup units are sometimes on standby, the summation of the power for all primary units gives an indication of the total power requirements for the astrionic equipment.

Tables 4-8 through 4-12 give the detailed weight and power breakdowns for the astrionic modules of the design space tug missions. Table 4-13 gives the summary of the astrionic module weights, and Table 4-14 gives the astrionic module power summary.

4.5 ASTRIONIC SYSTEM ELEMENTS LOCATED IN OTHER SPACE TUG MODULES

Some of the astrionic system elements identified during this study will also be located in other modules of the space tug. Figure 4-11 shows the interfaces between the astrionic module and other space tug modules. The following is a preliminary breakdown of the astrionic equipment in the propulsion module and the crew module. Follow-on study and/or other tug contractor studies will provide a detailed definition of these elements.

4.5.1 Preliminary Propulsion Module Astrionic Equipment

The only astrionic equipment identified for the propulsion module (PM) is standard interface units (SIU) to interface with the RCS and main thrust engines and the monitoring units which sequence and monitor equipment not serviced by other SIUs. These units will be serviced by the data bus network in the PM. The weight and power impacts for gimballed actuators for the main engine and for instrumentation were not addressed. Since the worst case mission identifies less than twenty of the above units, the astrionic weight impact would be less than 20 pounds and have an active power requirement of less than 30 watts. This impact should be minimal for the propulsion module.

4.5.2 Preliminary Crew Module Astrionic Equipment

The crew module (CM) will require SIUs, monitoring units for monitoring and sequencing of the crew module equipment and displays. These units will be serviced by the data bus network in the CM. As with the propulsion module, less than twenty units should be required, with less than twenty pounds of additional astrionic weight and less than thirty watts of active power requirements.

The display and control equipment will have a much greater impact on the crew module astrionic system weight and power. The following is a tabulation of the display and control equipment (See Appendix 0 for details) in the crew module and its associated weight and power.

Unit	Number Required	Unit Weight (Lbs)	Unit Power (Watts)	Total Weight (Lbs)	Total Power (Watts)
Cathode Ray Tubes	3	25	180	75	540
Keyboard	3	13	20	39	60
Manual Control Signal Connector	2	40	120	80	240
Totals				194	840

Some of the equipment presently located in the astrionic module (such as the laser radar sensor) may later be found to operate more efficiently in the crew module for manned missions. These components will operate satisfactorily on the astrionic module and are presently included in the astrionic module weight and power summaries.

Therefore, the weight and power for the astrionic equipment in the crew module is approximately 200 pounds with a power requirement of approximately 900 watts.

5.0 COMMONALITY OF SPACE TUG ASTRIONICS WITH OTHER SPACE SYSTEMS

5.1 INTRODUCTION AND SCOPE

A primary task in the Modular Astrionics Study is to: (1) identify and compare identical and related requirements, and (2) establish commonality in function, specifications, and software for the astrionic systems for the several planned future space systems. As part of this task, a comparison of the astrionic systems was made for:

- Space Tug (10 mission-vehicle combinations)
- Reusable Nuclear Shuttle (3 separate studies)
- Space Station
- Earth Orbital Shuttle (Orbiter and Booster)

All information used in this comparison was obtained for the six NASA contracted studies as listed in the references (Section 5.4).

5.2 SUMMARY

Considerable commonality exists in the recommended astrionic systems for the fourteen mission-vehicle combinations (10 tug, RNS, Space Station, Shuttle Booster, Shuttle Orbiter) at all levels. This commonality includes:

- Functions to be performed.
- Mechanization approach selected.
- Generic hardware recommended.

However, due to the level or phase of the studies, comparison of the performance requirements (specifications) for the subsystems or recommended hardware was not possible.

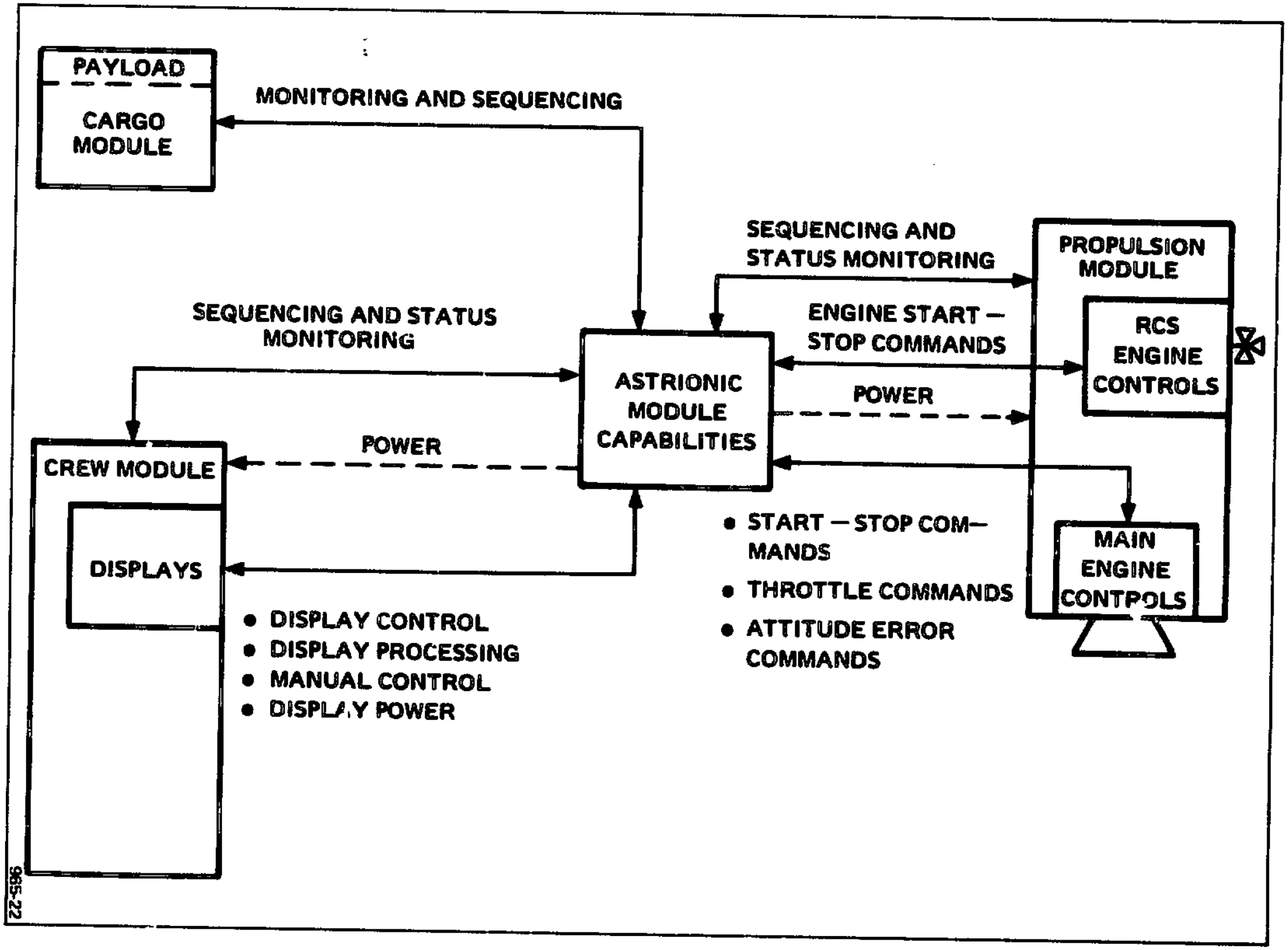

Figure 4-11. Space Tug Module Interfaces

Table 4-8. Detailed Weight and Power Summary – Synchronous Mission, Reusable Tugs (Sheet 1 of 2)

MISSION	SYNCHRONOUS (REUSABLE – 1ST TUG)				SYNCHRONOUS (REUSABLE – 2ND TUG)			
COMPONENT	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
CPU	1	1	80	120	1	1	80	120
BUS CONTROL UNIT	1	1	50	70	1	1	50	70
MAIN MEMORY	2		200	40	2		200	40
MAGNETIC TAPE	1		15	10	1		15	10
CONFIG. ASSIGN. UNIT	1	1	15	24	1	1	15	24
DATA MANAGEMENT SUBSYSTEM TOTALS			360	264			360	264
IMU	1		200	80	1		200	80
LASER RADAR	1		30	28	1		30	28
STAR TRACKER	2		40	46	2		40	46
LANDMARK TRACKER	1		40	30	1		40	30
HORIZON SENSOR	—		—	—	·		10	17
N, G&C SUBSYSTEM TOTALS			310	184			320	201
FUEL CELLS	2		200	338	2		200	338
REACTANT – H_2 (LBS)	—			2	—		—	11
REACTANT – O_2 (LBS)	—		—	11	—		—	89
H_2 TANK	2		—	2	2		—	13
O_2 TANK	2		—	3	2			27
DC REGULATOR	2		—	10	2		—	10
MOUNTING HARDWARE	—		—	20	—		—	20
ELEC. POWER SUBSYSTEM TOTALS			200	386			200	508
SIU	20	21	20	16.4	18	19	18	14.8
DATA BUS	1	2		22.5	1	2		22.5
MONITORING UNIT	9	9	18	9	9	9	18	9
POWER DISTRIBUTOR	1		—	39	1		—	39
AUX. POWER DISTRIBUTOR	4		—	52	4		—	52
WIRE & CABLES (LBS)	—		—	180	—		—	180
JUNCTION BOX	8		—	16	8		—	16
ELEC. NETWORKS SUBSYSTEM TOTALS			38	335			36	333
USB ELECTRONICS	1		107	42	1		107	42
USB DIPLEXER	1		—	0.5	1		—	0.5
USB ANTENNA SWITCH	1		—	0.3	1		—	0.3
USB POWER DIVIDER	1		—	0.5	1		—	0.5
USB OMNI ANTENNA	4		—	2	4		—	2
USB HI-GAIN ANT. CTRL	1		10	10	1		10	10
USB HI-GAIN ANTENNA	1		—	20	1		—	20
VHF DIPLEXER	1		—	1	1		--	1
VHF POWER DIVIDER	1		--	0.5	1		—	0.5
VHF OMNI ANTENNA	2		—	2.4	2		—	2.4
COMMAND DECODE ELEC.	2		10	20	2		10	20
TV CAMERA	1		6	5	1		6	5
TV CAMERA CTRL	1		3	2	1		3	2
VHF COMMAND RCVR.	2		2	4	2		2	4
CMND & CTRL SUBSYSTEM TOTALS			138	110			138	110

P = PRIMARY
B = BACKUP

Table 4-8. Detailed Weight and Power Summary – Synchronous Mission, Reusable Tugs (Sheet 2 of 2)

MISSION	SYNCHRONOUS (REUSABLE – 1ST TUG)				SYNCHRONOUS (REUSABLE – 2ND TUG)			
COMPONENT	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
	P	B			P	B		
COOLANT PUMP	1	1	150	50	1	1	150	50
SERVICE HEAT EXCH.	1		–	10	1		–	10
COOLANT ACCUM.	1		–	30	1		–	30
RADIATOR	8		–	147	8		–	147
COOLANT FLUID	–		–	50	–		–	50
LOUVERS	8		–	88	8		–	88
COMPONENT MTG. PANEL	8		–	200	8		–	200
MISC. PLUMBING	–		–	60	–		–	60
MULTILAYER INSULATION	–			25				25
THERMAL COND. SUBSYSTEM TOTALS			150	660			150	660
STRUCTURE (LBS)				400				400

P = PRIMARY
B = BACKUP

The more significant areas of commonality include the following:

- Prime Power Source – Fuel cells are recommended for all tug missions. Two of three studies recommend fuel cells for reusable nuclear shuttle (RNS), and fuel cells are recommended for the earth orbit shuttle (EOS) orbiter.

- Signal Distribution – All studies for all vehicles recommend use of a data bus system for internal signal distribution.

- Communications (RF) – All studies for all vehicles recommend unified S-band and VHF systems.

- Navigation and Guidance – Commonality in N & G hardware recommended included the following:

  1. IMU – All studies for all vehicles (except one RNS study) recommend the strapdown platform as the primary system for vehicle attitude and velocity data.

  2. Star Trackers – All studies for all vehicles (except EOS orbiter and booster) recommend use of star tracker(s) for navigation aids to obtain attitude and position updates.

  3. Landmark Tracker – The landmark tracker is recommended for all tug missions (except one), for the space station, and by one study for the RNS.

Table 4-9. Detailed Weight and Power Summary – Synchronous (Expendable Tug) and Lunar Landing Missions (Sheet 1 of 2)

MISSION	SYNCHRONOUS (EXPENDABLE TUG)				LUNAR LANDING			
COMPONENT	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
	P	B			P	B		
CPU	1		80	60	1	2	80	180
BUS CONTROL UNIT	1		50	35	1	2	50	105
MAIN MEMORY	1		100	20	2		200	40
MAGNETIC TAPE	—		—	—	1		15	10
DISPLAY MEMORY	—		—	—	1		50	10
CONFIG. ASSIGN. UNIT	—		—	—	1	2	15	36
AUX. MONT. UNIT	—		—	—	3		6	1.5
DATA MGMT. SUBSYSTEM TOTALS			230	115			416	383
IMU	1		200	80	1		200	80
LASER RADAR	1		30	28	1		30	28
STAR TRACKER	2		40	46	2		40	46
LANDMARK TRACKER	1		40	30	1		40	30
HORIZON SENSOR	1		10	17	—		—	—
LANDING RADAR	—		—	—	1		80	34
N, G&C SUBSYSTEM TOTALS			320	201			390	218
FUEL CELL	1		100	169	2		200	338
REACTANT – H_2	—			3	—			5
REACTANT – O_2	—			17	—			43
H_2 TANK	1			3	2			6
O_2 TANK	1			5	2			13
BATTERY	—			—	—	1		140
DC REGULATOR	1			5	2			10
BATTERY CHARGER	—			—	1			8
MOUNTING HARDWARE	—		—	20	—			20
ELEC. POWER SUBSYSTEM TOTALS			100	222			200	583
SIU	12		12	4.8	23	22	23	18
DATA BUS	1		—	7.5	1	2	—	22.5
MONITORING UNIT	9		18	4.5	9	9	18	9
AUX. MONT. UNIT	—		—	—				
POWER DISTRIBUTOR	1		—	39	1		—	39
AUX. POWER DISTR.	4		—	52	4		—	52
WIRE & CABLES (LBS)	—		—	150	—		—	200
JUNCTION BOX	8		—	16	8		—	16
ELEC. NETWORKS SUBSYSTEM TOTALS			30	274			41	356
USB ELECTRONICS	1		107	42	1		107	42
USB DIPLEXER	1		—	0.5	1		—	0.5
USB ANTENNA SWITCH	1		—	0.3	1		—	0.3
USB POWER DIVIDER	1		—	0.5	1		—	0.5
USB OMNI ANTENNA	4		—	2	4		—	2
USB HI-GAIN ANT. CTRL	1		10	10	1		10	10
USB HI-GAIN ANT.	1		—	20	1		—	20
VHF TRANSCEIVER					1		25	15
VHF DIPLEXER					1			1
VHF POWER DIVIDER					1			0.5
VHF OMNI ANT.					2			2.4
COMMAND DECODE ELEC.	1		5	10	1	1	5	20
TV CAMERA	1				1		6	5
TV CAMERA CTRL	1		—	—	1		3	2
AUDIO EQUIPMENT	—				1	—	3	2
CMND & CTRL SUBSYSTEM TOTALS			122	87			159	125

P = PRIMARY
B = BACKUP

Table 4-9. Detailed Weight and Power Summary – Synchronous (Expendable Tug) and Lunar Landing Missions (Sheet 2 of 2)

MISSION	SYNCHRONOUS (EXPENDABLE TUG)				LUNAR LANDING			
COMPONENT	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
COOLANT PUMP	1	1	150	50			TO BE DETERMINED	
SERVICE HEAT EXCH.	1		—	10				
COOLANT ACCUM.	1		—	30				
RADIATOR	6		—	110				
COOLANT FLUID	—		—	50				
LOUVERS	6		—	66				
COMP. MTG. PANEL	6		—	150				
MISC. PLUMBING	—		—	60				
MULTILAYER INSULATION	—		—	25				
THERMAL COND. SUBSYSTEM TOTALS			150	551	ESTIMATES:		150	1000
STRUCTURES (LBS)				400				400

P = PRIMARY
B = BACKUP

4. Laser Radar – For missions requiring rendezvous and docking operations, all studies for all vehicles recommend use of the laser radar, except those studies which assigned docking as a crew function.

5.3 COMPARISONS FOR COMMONALITY

Comparisons to identify commonality in the recommended astrionic systems for fourteen mission-vehicle combinations were made using the information available from the appendices to this report and the documents listed in the references. These comparisons were made at four study levels and are discussed in the following paragraphs.

5.3.1 Missions and Mission Elements

The areas for comparison at this level are:

- Vehicles – Whether manned or automated and reusable or expended.
- Operational Base – Location where the "prepare for mission" function is performed.
- Space Operations – Assembly and/or maintenance operations to be performed in space.

Table 4-10. Detailed Weight and Power Summary – Earth Orbital and Lunar Orbital Operations Missions (Sheet 1 of 2)

MISSION	EARTH ORBITAL OPERATIONS				LUNAR ORBITAL OPERATIONS			
COMPONENT	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
	P	B			P	B		
CPU	1	2	80	180	1	2	80	180
BUS CONTROL UNIT	1	2	50	105	1	2	50	105
MAIN MEMORY	2		200	40	2		200	40
MAGNETIC TAPE UNIT	1		15	10	1		15	10
DISPLAY MEMORY	1		50	10	1		50	10
CONFIG. ASSIGNMENT UNIT	1	2	15	36	1	2	15	36
DATA MGMT. SUBSYSTEM TOTALS			410	381			410	381
IMU	1		200	80	1		200	80
LASER RADAR	1		30	28	1		30	28
STAR TRACKER	2		40	46	2		40	46
LANDMARK TRACKER	1		40	30	1		40	30
N, G&C SUBSYSTEM TOTALS			310	184			310	184
FUEL CELL	2		200	338	2		200	338
REACTANT – H_2	–		–	4	–		–	4
REACTANT – O_2	–		–	36	–		–	36
H_2 TANK	2		–	5	2		–	5
O_2 TANK	2		–	11	2		–	11
BATTERY		1	–	140		1	–	140
DC REGULATOR	2		–	10	2		–	10
BATTERY CHARGER	1		–	8	1		–	8
ELEC. POWER SUBSYSTEM TOTALS			200	552			200	552
SIU	22	21	22	17.2	22	21	22	17.2
DATA BUS	1	2	–	22.5	1	2	–	22.5
MONITORING UNIT	9	9	18	9	9	9	18	9
POWER DISTRIBUTOR	1		–	39	1		–	39
AUX. POWER DISTRIBUTOR	4	1	–	65	4	1	–	65
WIRE & CABLES			–	200			–	200
JUNCTION BOXES	8		–	16	8		–	16
ELEC. NETWORKS SUBSYSTEM TOTALS			40	369			40	369
USB ELECTRONICS	1		107	42	1		107	42
USB DIPLEXER	1		–	0.5	1		–	0.5
USB ANTENNA SWITCH	1		–	0.3	1		–	0.3
USB POWER DIVIDER	1		–	0.5	1		–	0.5
USB OMNI ANTENNAS	4		–	2	4		–	2
USB HI-GAIN ANT. CTRL	1		10	10	1		10	10
USB HI-GAIN ANTENNAS	1		–	20	1		–	20
VHF TRANSCEIVER	1		25	15	1		25	15
VHF DIPLEXER	1		–	1	1		–	1
VHF POWER DIVIDER	1		–	0.5	1		–	0.5
VHF OMNI ANTENNAS	2		–	2.4	2		–	2.4
CMD. DECODE ELECTRONICS	2		10	20	1	1	5	20
TV CAMERA	1		6	5	1		6	5
TV CAMERA CTRL	1		3	2	1		3	2
VHF COMMAND RCVR	2		2	4	–		–	–
AUDIO EQUIPMENT	1		3	2	1		3	2
CMND & CTRL SUBSYSTEM TOTALS			166	127			159	123

P = PRIMARY
B = BACKUP

Table 4-10. Detailed Weight and Power Summary – Earth Orbital and Lunar Orbital Operations Missions (Sheet 2 of 2)

MISSION	EARTH ORBITAL OPERATIONS				LUNAR ORBITAL OPERATIONS			
COMPONENT	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
COOLANT PUMP	1	1	150	50	1	1	150	50
SERVICE HEAT EXCHANGER	1		–	10	1		–	10
COOLANT ACCUMULATOR	1		–	30	1		–	30
RADIATOR	8		–	147	8		–	147
COOLANT FLUID	–		–	50	–		–	50
LOUVERS	8		–	88	8		–	88
COMPONENT MTG. PANEL	8		–	200	8		–	200
MISCELLANEOUS PLUMBING	–		–	60	–		–	60
MULTILAYER INSULATION	–		–	25			–	25
THERMAL COND. SUBSYSTEM TOTALS			150	660			150	660
STRUCTURES (LBS)	–		–	400	–		–	400

P = PRIMARY
B = BACKUP

- Interfaces – RF interfaces with other space systems and ground stations.
- Transport System – Transport system used, if required, to transport the vehicle to its operating base.

Comparisons in these areas are illustrated in Figure 5-1 which shows both the commonalities and differences. Significant areas of commonality are:

- In-space maintenance is required for RNS, tug, and space station.
- Signal interfaces (RF) are required between the space systems and between the space systems and the ground stations.
- Automated missions (unmanned) are required for both RNS and tug.

5.3.2 Mission Events

The fourteen missions were categorized into eleven mission events. Comparisons of these mission events are illustrated in Figure 5-2. Considerable commonality exists for the intraorbit transfers, rendezvous and dock, and station keeping events. Commonality exists between at least two space vehicles for each of the eleven mission events except:

- Lunar parking orbit (LPO) to low earth orbit (LEO) transfers are a requirement for RNS only. However, the astrionic system functions and their mechanization for providing N & G are essentially the same for LPO to LEO transfers as those for LEO to LPO transfers, thus providing commonality between two vehicles for these mission events.

Table 4-11. Detailed Weight and Power Summary – Planetary Mission

MISSION	PLANETARY (REUSEABLE – 1ST TUG)				PLANETARY (EXPENDABLE – 2ND TUG)			
COMPONENT	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY		TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
	P	B			P	B		
CPU	1	1	80	120	1	1	80	120
BUS CONTROL UNIT	1	1	50	70	1	1	50	70
MAIN MEMORY	2		200	40	2		200	40
MAGNETIC TAPE UNIT	1		15	10	1		15	10
CONFIG. ASSIGN. UNIT	1	1	15	24	1	1	15	24
DATA MGMT. SUBSYSTEM TOTALS			360	264			360	264
IMU	1		200	80	1		200	80
LASER RADAR	1		30	28	—		—	—
STAR TRACKER	2		40	46	2		40	45
LANDMARK TRACKER	1		40	30	—		—	—
HORIZON SENSOR	1		10	17	—		—	—
N, G&C SUBSYSTEM TOTALS			320	201			240	126
FUEL CELL	2		200	338	1		100	169
REACTANT – H_2	—		—	4	—		—	1
REACTANT – O_2	—		—	36	—		—	8
H_2 TANK	2		—	5	1		—	1
O_2 TANK	2		—	11	1		—	3
DC REGULATOR	2		—	10	1		—	10
ELEC. POWER SUBSYSTEM TOTALS			200	404			100	192
SIU	23	22	23	18	22	21	22	17.2
DATA BUS	1	2	—	22.5	1	2	—	22.5
MONITORING UNIT	9	9	18	9		9	18	9
POWER DISTRIBUTOR	1		—	39			—	39
AUX. POWER DISTRIBUTOR	4		—	52			—	52
WIRES & CABLES (LBS)	—		—	180	—		—	150
JUNCTION BOX	8		—	16	8		—	16
ELEC. NETWORKS SUBSYSTEM TOTALS			41	336			40	306
USB ELECTRONICS	1		107	42	1		107	42
USB DIPLEXER	1		—	0.5	1		—	0.5
USB ANTENNA SWITCH	1		—	0.3	1		—	0.3
USB POWER DIVIDER	1		—	0.5	1		—	0.5
USB OMNI ANTENNAS	4		—	2	4		—	2
USB HI-GAIN ANT. CTRL	1		10	10	1		10	10
USB HI-GAIN ANTENNA	1		—	20	1		—	20
COMMAND DECODE ELEC.	1	1	5	20	1	1	5	20
TV CAMERA	1		6	5	1		6	5
TV CAMERA CONTROL	1		3	2	1		3	2
CMND & CTRL SUBSYSTEM TOTALS			131	102			131	102
COOLANT PUMP	1	1	150	50	1	1	150	50
SERVICE HEAT EXCHANGER	1		—	10	1		—	10
COOLANT ACCUMULATOR	1		—	30	1		—	30
COOLANT FLUID	—		—	50	—		—	50
RADIATOR	8		—	147	6		—	110
LOUVERS	8		—	88	6		—	66
COMP. MTG. PANEL	8		—	200	6		—	150
MULTILAYER INSULATION	—		—	25	—		—	25
MISC. PLUMBING	—		—	60	—		—	60
THERMAL COND. SUBSYSTEM TOTALS			150	660			150	551
STRUCTURES (LBS)	—		—	400	—			400

P = PRIMARY
B = BACKUP

Table 4-12. Detailed Weight and Power Summary – Reusable Nuclear Shuttle and Four Stage Saturn V Missions (Sheet 1 of 2)

MISSION	REUSABLE NUCLEAR SHUTTLE				FOUR STAGE SATURN V			
COMPONENT	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
CPU	1	1	80	120	1	1	80	120
BUS CONTROL UNIT	1	1	50	70	1	1	50	70
MAIN MEMORY	2		200	40	2		200	40
MAGNETIC TAPE UNIT	1		15	10	1		15	10
CONFIG. ASSIGN. UNIT	1	1	15	24	1	1	15	24
DATA MGMT. SUBSYSTEM TOTALS			360	264			360	264
IMU	1		200	80	1		200	80
LASER RADAR	1		30	28	1		30	28
STAR TRACKER	2		40	46	2		40	46
LANDMARK TRACKER	1		40	30	1		40	30
HORIZON SENSOR	1		10	17	1		10	17
RATE GYROS	–		–	–	1		45	11
ACCELEROMETERS	–		–	–	1		8.4	4
N, G&C SUBSYSTEM TOTALS			320	201			373	216
FUEL CELL	2		200	338	2		200	338
REACTANT – H_2	–		–	54	–		–	17
REACTANT – O_2	–		–	486	–		–	161
H_2 TANK	2		–	72	2		–	22
O_2 TANK	2		–	144	2		–	45
BATTERY	2		–	156	–		–	–
DC REGULATOR	2		–	10	2		–	10
BATTERY CHARGER	1		–	8	–		–	–
ELEC. POWER SUBSYSTEM TOTALS			200	1268			200	593
SIU	23	22	23	18	23	22	23	18
DATA BUS	1	2	–	22.5	1	2	–	22.5
MONITORING UNIT	9	9	18	9	9	9	18	9
POWER DISTRIBUTOR	1		–	39	1		–	39
AUX. POWER DISTRIBUTOR	4	1	–	65	4		–	52
WIRES & CABLES	–		–	200	–		–	180
JUNCTION BOX	8		–	16	8		–	16
ELEC. NETWORKS SUBSYSTEM TOTALS			41	369			41	336
USB ELECTRONICS	1		107	42	1		107	42
USB DIPLEXER	1		–	0.5	1		–	0.5
USB ANTENNA SWITCH	1		–	0.3	1		–	0.3
USB POWER DIVIDER	1		–	0.5	1		–	0.5
USB OMNI ANTENNAS	4		–	2	4		–	2
USB HI-GAIN ANT. CTRL	1		10	10	1		10	10
USB HI-GAIN ANTENNA	1		–	20	1		–	20
COMMAND DECODE ELECT.	1	1	5	20	1	1	5	20
TV CAMERA	1		6	5	1		6	5
TV CAMERA CONTROL	1		3	2	1		3	2
CMND & CTRL SUBSYSTEM TOTALS			131	102			131	102

P = PRIMARY
B = BACKUP

Table 4-12. Detailed Weight and Power Summary – Reusable Nuclear Shuttle and Four Stage Saturn V Missions (Sheet 2 of 2)

MISSION	REUSABLE NUCLEAR SHUTTLE				FOUR STAGE SATURN V			
COMPONENT	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)	QUANTITY P	QUANTITY B	TOTAL POWER (WATTS)	TOTAL WEIGHT (LBS)
COOLANT PUMP	1	1	150	50	1	1	150	50
SERVICE HEAT EXCH.	1			10	1			10
COOLANT ACCUMULATOR	1			30	1			30
COOLANT FLUID	—			50	—			50
RADIATOR	8			147	8			147
LOUVER	8			88	8			88
COMPONENT MTG. PANEL	8			200	8			200
MULTILAYER INSULATION	—			25	—			25
MISC. PLUMBING	—			60	—			60
THERMAL COND. SUBSYSTEM TOTALS			150	660			150	660
STRUCTURES (LBS)	—		—	400	—		—	400

P = PRIMARY
B = BACKUP

- Lunar landing is a requirement for the tug only. This event establishes several special astrionic requirements, including:

  1. Control of variable thrust engines.

  2. Use of a landing radar system to provide terminal guidance data (at unprepared sites).

  3. Perform the power-up, checkout and system initialization functions without any direct support from an operational base.

5.3.3 Astrionic Functional Areas

The astrionic systems were broken down into nine functional areas for comparison as illustrated in Figure 5-3. Considerable commonality exists in all functional areas at this level. The more significant areas of commonality and differences are as follows:

- Navigation and Guidance – All studies for all mission-vehicle combinations (except one tug mission) recommend use of an inertial and terminal guidance phase. The inertial navigation, as recommended by the studies, is performed on the basis of measured vehicle attitude and accelerations in the inertial measurement unit and then processed in the digital computer to obtain vehicle

Table 4-13. Astrionic Module Weight Summaries

SUBSYSTEM	MISSION: SYNCHRONOUS (EXPENDABLE TUG)	SYNCHRONOUS (REUSABLE 1ST TUG)	SYNCHRONOUS (REUSABLE 2ND TUG)	LUNAR LANDING	EARTH ORBIT OPERATIONS	LUNAR ORBIT OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	REUSABLE NUCLEAR SHUTTLE	FOUR STAGE SATURN V
DATA MANAGEMENT	115	284	284	383	381	381	284	284	284	284
NAV., GUID. AND CONTROL	201	184	201	219	184	184	201	128	201	218
ELECTRICAL POWER	222	386	583	583	582	582	404	182	1288	683
ELECTRICAL NETWORKS	274	336	333	356	388	388	335	388	388	336
COMMAND AND CONTROL	87	110	110	125	127	123	102	102	182	102
INSTRUMENTATION*	40	50	50	50	50	5C	50	40	50	50
THERMAL CONDITIONING	581	680	680	1080*	680	680	680	581	680	680
SUBTOTALS	1480	1980	2128	2715	2323	2319	2017	1980	2814	2221
STRUCTURES**	400	400	400	400	400	400	400	400	400	400
TOTALS	1880	2380	2528	3115	2723	2719	2417	2380	3314	2621

*ESTIMATED WEIGHTS
**EXPECTED WEIGHT. DEPENDENT UPON SELECTED STRUCTURAL DESIGN AND REQUIRED MICROMETEROID PROTECTION

NOTE WEIGHT AS GIVEN IN POUNDS

Table 4-14. Astrionic Module Power Summaries

SUBSYSTEM	MISSION: SYNCHRONOUS (EXPENDABLE TUG)	SYNCHRONOUS (REUSABLE 1ST TUG)	SYNCHRONOUS (REUSABLE 2ND TUG)	LUNAR LANDING	EARTH ORBIT OPERATIONS	LUNAR ORBIT OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	REUSABLE NUCLEAR SHUTTLE	FOUR STAGE SATURN V
DATA MANAGEMENT	230	380	380	418	410	410	380	380	380	380
NAV., GUID. AND CONTROL	320	310	320	380	310	310	320	240	320	373
ELECTRICAL POWER**	100	200	200	200	200	200	200	100	200	200
ELECTRICAL NETWORKS	30	38	36	41	40	40	41	40	41	41
COMMAND AND CONTROL	122	138	138	158	186	159	131	131	131	131
THERMAL CONDITIONING	150	150	150	150*	150	150	150	150	150	150
TOTALS	982	1198	1204	1358	1276	1288	1202	1021	1202	1258

*ESTIMATED POWER
**FUEL CELL PARASITIC POWER

NOTE: POWER GIVEN IN WATTS

	TUG MISSION VEHICLE COMBINATIONS										RNS			EOS		SS	NOTES
	SYNC ORBIT (EXPENDABLE VEHICLE)	SYNC ORBIT (REUSABLE 1ST TUG)	SYNC ORBIT (REUSABLE 2ND TUG)	LUNAR LANDING	EARTH ORBITAL OPERATIONS	LUNAR ORBITAL OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	FOUR STAGE SATURN V	RNS USING TUG ASTRIONICS MODULE	LOCKHEED STUDY REPORT	NAR STUDY REPORT	MDAC STUDY REPORT	ORBITER	BOOSTER	SPACE STATION	
VEHICLE																	
– MANNED				X	X	X				X	(1)	(1)	(1)	X	X	X	(1) STUDY GUIDELINE HAS MAN AS A NON-PARTICIPANT
– AUTOMATED	X	X	X	X	X	X	X	X	X	X	X	X	X				
– REUSABLE		X	X	X	X	X	X				X	X	X	X	X	–	(2) DEPENDENT ON TRANSPORT SYSTEM USED
– EXPENDED (SINGLE MISSION)	X							X								–	
OPERATIONAL BASE																	(3) EITHER OR BOTH TRANSPORT SYSTEMS MAY BE USED
– EARTH	X								X					X	X		
– SPACE STATION BASE		X	X	X	X	X	X	X		X	X	X	X			–	
SPACE OPERATIONS																	
– ASSEMBLY		X	X				X	X		X(2)						X	
– MAINTENANCE		X	X	X	X	X	X	X		X	X	X	X			X	
INTERFACES (SIGNAL)																	
– GROUND STATIONS	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	
– OTHER SPACE STATIONS	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	
TRANSPORT SYSTEM																	
– SATURN DERIVATIVE				X		X				X(3)						X	
– EOS	X	X	X	X	X	X	X	X		X(3)						X	
– RNS				X		X											

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Figure 5-1. Missions and Mission Elements Commonality

	TUG MISSION-VEHICLE COMBINATIONS										RNS			EOS		SS
	SYNC ORBIT (EXPENDABLE VEHICLE)	SYNC ORBIT (REUSABLE 1ST TUG)	SYNC ORBIT (REUSABLE 2ND TUG)	LUNAR LANDING	EARTH ORBITAL OPERATIONS	LUNAR ORBITAL OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	FOUR STAGE SATURN V	RNS - USING TUG ASTRIONICS MODULE	LOCKHEED STUDY REPORT	NAR STUDY REPORT	MDAC STUDY REPORT	ORBITER	BOOSTER	SPACE STATION
POWERED ASCENT									X			X(2)	X(2)	X	X	
INTRAORBIT TRANSFERS		X	X	X	X	X	X			X				X		
LEO TO LPO TRANSFER									X	X	X	X	X			
LPO TO LEO TRANSFER										X	X	X	X			
LEO TO SYNC. ORBIT	X		X							(4)	X	X	X			
SYNC. ORBIT TO LEO			X							(4)	X	X	X			
RENDEZVOUS AND DOCK	X	X	X	X	X	X	X			X				X		X(1)
STATION KEEPING	X		X		X	X				X				X		X
LEO TO DEEP SPACE								X								
LUNAR LANDING AND RETURN				X												
RE-ENTRY AND LAND LANDING														X	X	

NOTES

(1) REMOTE CONTROL FOR CHASER VEHICLE
(2) PROVIDES N, G&C FOR INT. 21 VEHICLE
(3) ONLY PRIMARY RNS MISSIONS SHOWN
(4) NOT INCLUDED IN BASELINE MISSIONS SELECTED FOR STUDY

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Figure 5-2. Mission Events Commonality

	TUG MISSION-VEHICLE COMBINATIONS									RNS			EOS		SS
	SYNC. ORBIT (EXPENDABLE VEHICLE)	SYNC. ORBIT (REUSABLE 1ST TUG)	SYNC. ORBIT (REUSABLE 2ND TUG)	LUNAR LANDING	EARTH ORBITAL OPERATIONS	LUNAR ORBITAL OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	FOUR STAGE SATURN V	RNS – USING TUG ASTRIONICS MODULE	LOCKHEED STUDY REPORT	NAR STUDY REPORT	MDAC STUDY REPORT	ORBITER	BOOSTER	SPACE STATION
NAVIGATION AND GUIDANCE																
– INERTIAL	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X(1)
– TERMINAL	X	X	X	X	X	X	X		X(6)	X	X	X	X	X	X	X(2)
CONTROL																
– MAIN ENGINE(S) (GIMBALS)	X	X	X	X	X	X	X	X	X	X(3)	X(3)	X(3)	X(3)	X	X	
– THRUSTERS (APS)	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
– CMG'S																X
DATA MANAGEMENT																
– CPU	X	X	X	X	X	X	X	X	X	X	X	N/A	X			
– OTHER (MULTIPROCESSORS)												N/A		X	X	X
COMMUNICATIONS																
– VHF		X	X	X	X	X					X	X	X	X	X	X
– UNIFIED S-BAND	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
– K_U BAND																X
– OTHER												X(4)		X(5)	X(5)	
PRIME POWER																
– CENTRAL											X	X	X	X	X	X
– DISTRIBUTED	X	X	X	X	X	X	X	X	X	X						
THERMAL CONTROL																
– ACTIVE	X	X	X	X	X	X	X	X	X	X	N/A	N/A	N/A	X	X	X
– PASSIVE											N/A	N/A	N/A			
– INTEGRATED WITH LIFE SUPPORT											N/A	N/A	N/A	X	X	X
– COMPONENT TCS	X	X	X	X	X	X	X	X	X	X	N/A	N/A	N/A			
STRUCTURE AND PACKAGING																
– DISTRIBUTED												N/A		X	X	X
– MODULE STRUCTURE	X	X	X	X	X	X	X	X	X	X		N/A	X			
– MODULE PACKAGE	X	X	X	X	X	X	X	X	X	X	X	N/A	X	N/A	N/A	N/A
CREW DISPLAYS AND CONTROLS	X	X	X	X	X	X	X	X	X	X	N/A	N/A	N/A	X	X	X
CHECKOUT AND MONITOR	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X

NOTES

(1) STATION KEEPING FUNCTION
(2) REMOTE CONTROL OF CHASER VEHICLE FOR RENDEZVOUS AND DOCK.
(3) SPECIAL NUCLEAR ENGINE CONTROLS REQUIRED
(4) HF AND UHF SYSTEMS
(5) VOR AND DME FOR NAVIGATION
(6) LUNAR APPROACH PHASE
N/A DENOTES INFORMATION NOT AVAILABLE

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Figure 5-3. Astrionic Functional Area Commonality

navigation data from the solution of the vehicle's equations of motion. According to the selected guidance law, the guidance computer generates vehicle guidance command signals. Navigation aids (star tracker, landmark tracker, etc.) are provided for IMU realignment, position and/or state vector updates. Terminal guidance utilizes an on-board sensor(s) to provide the vehicle-to-target relationship in real time. Again according to the selected guidance law and using this real time data, the guidance computer (except where assigned as a crew function) generates vehicle guidance command signals. The terminal guidance phase is used for the rendezvous, docking, planet approach, lunar and earth landing, as required by the individual missions.

The space station has an inertial guidance requirement which is primarily a station keeping function. The terminal guidance provides a remote control capability for traffic control, rendezvous and docking.

- Data Management – All studies recommend a data management system utilizing digital computer(s) to provide on-board process control and computations for a family of functions. This family of functions for most mission-vehicles include:

  1. Navigation and IMU processing
  2. Guidance
  3. Control
  4. Sequencing
  5. Data bus control
  6. Monitor and checkout
  7. Telemetry
  8. Display support (crew)
  9. Software management

  The studies (which recommended a computer approach) recommend a central processor for tug and RNS. Multiprocessors were recommended for the EOS and space station.

- Communications – All studies for all missions recommend unified S-band systems. VHF systems were recommended, in addition to the S-band systems, for RNS, EOS, space station, and for the manned tug missions.

- Other Areas – At this level of comparison, two significant differences develop due to the tug modularity requirement. These are:

  1. Each tug module must contain its own prime power source and distribution system where the EOS, RNS, and space station have central prime power sources.

2. The tug requires a component thermal conditioning system where the EOS and space station studies recommend an integrated thermal control and life support system. All studies that addressed the thermal control problem recommended an active system.

5.3.4 Subsystem Baseline Hardware

Three subsystems (power, navigation and guidance, and signal distribution) were selected to illustrate commonality at the hardware level. However, it should be noted that this comparison is made as to generic hardware types and not according to specific requirements or specifications. The commonality in hardware usage for these three selected subsystems is illustrated in Figure 5-4. As shown in Figure 5-4, some commonality exists in nearly all areas. The more significant areas of commonality and differences are as follows:

- Prime Power – All studies for all systems (except one RNS study and for the space station) recommend use of fuel cells, batteries or a combination of fuel cells and batteries. An Isotope Brayton system was recommended for the space station due to its large (~25 KWE) power load and long mission duration.

- Navigation and Guidance -

  1. IMU – All studies for all systems (except one RNS study) recommend use of a strapdown IMU (with digital processor) as the primary inertial system for measuring vehicle rates and accelerations. One RNS study recommended a gimbaled IMU in a "strapdown-to-the-stars" configuration.

  2. Computer – All studies for all systems recommended use of a digital computer (either dedicated or use of a central process unit) for performing the N & G processing and computations.

  3. Star Trackers – All studies for all vehicles (except EOS orbiter and booster) recommend use of star trackers for navigation aids to obtain attitude and position updates. The EOS orbiter and booster are manned vehicles and utilize a sextant for attitude and position updates.

     The space station uses both star trackers and star sensors. The star trackers are used when the space station is in an inertial hold position (fixed with respect to the stars). The star trackers are locked-on to selected stars using gimbals and measure the star angles in vehicle coordinates. The star sensors are used when the space station is in an earth fixed position (rotating in an inertial coordinate system). The star sensors provide transition time for selected stars. Correlation of the knowledge of star celestial coordinates with the transition time provides partial information on vehicle inertial attitude. Multiple transits on different stars provide the necessary data for complete attitude update.

BASELINE HARDWARE	SPACE TUG	RNS: LMSC REPORT	RNS: NAR REPORT	RNS: MDAC REPORT	EARTH ORBITAL SHUTTLE	SPACE STATION
ELECTRICAL POWER						
– BATTERIES	●	●	●	●	●	
– FUEL CELLS	●		●	●	● (1)	
– SOLAR ARRAYS		●				
– ISOTOPE BRAYTON						● (2)
DATA MANAGEMENT						
– DIGITAL COMPUTERS (3)	●	●	●	●	●	●
NAVIGATION, GUIDANCE AND CONTROL						
– IMU, STRAPDOWN	●	●	●	●	●	●
– IMU, GIMBAL			●			
– STAR TRACKER(S)	●	●	●	●	●	●
– STAR SENSOR						● (4)
– LANDMARK TRACKER	●			●		●
– SEXTANT	(5)				● (6)	
– HORIZON SENSOR	●	●	●	●		●
– RADAR (RF)			●			●
– LASER RADAR	●	●	(7)	●	(7)	●
– LANDING RADAR	● (8)					
– RADAR ALTIMETER		●	●		●	
– SUN SENSOR		●				
– ILS AND MARKER BEACON					● (9)	
– VOR					● (9)	
– DME					● (9)	
ELECTRICAL NETWORKS						
– DATA BUS	●	●	● (10)	●	●	●

NOTES

(1) ORBITER ONLY
(2) LARGE (~25 KWE) POWER REQUIREMENT
(3) EITHER DEDICATED OR USE OF CENTRAL UNIT
(4) FOR USE DURING EARTH FIXED ATTITUDE POSITIONS
(5) FOR MANNED MISSIONS; MAY BE USED AS BACKUP
(6) EOS MISSIONS ARE MANNED
(7) GUIDANCE FOR DOCKING ASSIGNED AS CREW FUNCTION
(8) FOR LUNAR LANDING MISSION
(9) REQUIRED FOR LANDING AT PREPARED LANDING STRIP
(10) CALLED PARTY LINE SYSTEM

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Figure 5-4. Selected Subsystems Baseline Hardware Commonality

4. Landmark Trackers – The landmark tracker is recommended for all tug missions (except one), for the space station, and by one RNS study. From the referenced studies and from the sensors and combinations of sensors investigated, the star trackers (two) and landmark tracker combinations provided the most accurate navigation data. However, the landmark tracker's use is limited in altitude and most studies for high altitude or long range operations added a horizon sensor or sun sensor to replace the landmark tracker. The EOS orbiter and booster use a sextant which can be used as a landmark tracker.

5. Laser Radar – For missions requiring rendezvous and docking, all studies for all vehicles recommend use of a laser radar except those studies where docking was assigned as a crew function.

6. Instrument Landing System (ILS) and Marker Beacon, VHF Omni Ranging (VOR), and Distance Measuring Equipment (DME) – These systems are to provide navigation and guidance data for land landings. Only the EOS orbiter and booster have earth land landing requirements.

- Signal Distribution – All studies for all vehicles recommend use of a data bus system for internal signal distribution.

The requirements (memory and operational speed) for digital computer(s) are shown in Table 5-1. As shown in the table, the space station requirements exceed all other mission-vehicle requirements by almost an order of magnitude. The range of requirements for mission-vehicles numbered (2) through (13) is 176,000 to 352,000 operations per second and 31,500 to 73,600 32-bit words of main memory. A "common" CPU and main storage device could be developed for the spectrum of future space vehicles, and multiples of a "common" system (1, 2, 3 etc.) could be used for vehicles having requirements larger than the "common" system could provide.

5.4 REFERENCES

The following were used as references for the data in this section of the report:

- All appendices to this report.

- North American Rockwell, Space Division Nuclear Flight Systems Definition Study, dated 19 May 1970.

  1. Final Phase II Review Document.

  2. Vol. IV of Final Report, pages 6-1 through 6-137.

- McDonnell Douglas Astronautics Company, Nuclear Flight Systems Definition Study, dated May 1970.

  1. Phase II Final Briefing Document.

  2. Vol. II Part I of Final Report, pages 214 through 301

Table 5-1. CPU and Main Storage Requirements

MISSION-VEHICLE	OPERATIONS PER SECOND (1)	MAIN STORAGE (32 BIT WORDS)
SPACE STATION	1,132,000	681,000
SPACE SHUTTLE BOOSTER	352,000	73,600
SPACE SHUTTLE ORBITER	352,000	70,000
TUG – SYNC. ORBIT (EXPENDABLE)	153,000	32,500
TUG – SYNC. ORBIT (REUSABLE 1ST TUG)	176,000	45,000
TUG – SYNC. ORBIT (REUSABLE 2ND TUG)	176,000	45,000
TUG – LUNAR LANDING	278,000	60,000
TUG – EARTH ORBIT OPERATIONS	272,000	59,000
TUG – LUNAR ORBIT OPERATIONS	272,000	59,000
TUG – PLANETARY (REUSABLE 1ST TUG)	170,000	39,500
TUG – PLANETARY (EXPENDABLE 2ND TUG)	170,000	31,500
TUG – RNS	278,000(2)	63,000(2)
TUG – FOUR STAGE SATURN V	275,000	60,500
RNS – (MDAC, STUDY REPORT)	250,000(3)	38,000(3)
RNS – (LMSC, STUDY REPORT)	N/A	79,500
RNS – (NAR, STUDY REPORT)	N/A	N/A

NOTES:
(1) MAXIMUM RATE
(2) DOES NOT INCLUDE NUCLEAR ENGINE CONTROL
(3) AUTONOMOUS N&G REQUIREMENTS ONLY

- Lockheed Missiles and Space Company, Nuclear Flight Systems Definition Study, dated May 1970.
  1. Final Briefing Document.
  2. Vol. III of Final Report, pages 7-1 through 7-60.
- McDonnell Douglas Astronautics Company, Space Station Definition, MDAC G0605, dated July 1970, Vol. V Books 1 through 5.
- North American Rockwell Space Division, Proposal for Phase B Space Shuttle Program, dated 27 March 1970.

6.0 RECOMMENDATIONS FOR FUTURE STUDY

The study effort to date has concentrated on the preliminary design of a modular astrionic system (which is packaged in a structural module called the astrionic module) to meet the requirements of the space tug missions. Emphasis has been placed on the design and physical definition of the total system as well as its associated components, i.e. modular astrionics.

As additional information on the space tug propulsion and crew modules becomes available, the astrionic system design must be iterated upon. This section recommends future study effort for some of the astrionic subsystems as well as the integration effort with other space tug modules.

6.1 GENERAL STUDIES

There are several general areas of study effort required for the space tug astrionics. These areas are as follows:

- Correlate study results with astrionic system studies currently being conducted by NASA on space tug, space shuttle, reusable nuclear shuttle and space station vehicles for investigation of commonality of functional requirements and for compatibility of electrical, mechanical, and operational timeline interfaces.

- Iterate on the design of the astrionic systems for the design space tug missions. This will include better definition of subsystem components based on the preliminary definition of the astrionic system and its components. Particular attention will be paid to the impact of commonality across tug missions.

- Continue the requirements analysis for more explicit definition of the functional requirements and specifications for the space tug astrionics. These requirements are to be continually factored into the design of the astrionic system.

- Perform cost-effectiveness trade studies (see "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974," Progress Report – December 16, 1969 to February 15, 1970, IBM No. 69-K44-0006C for details concerning cost-effectiveness analysis) to determine cost-effective equipment usage and to provide cost-effective designs for the astrionic system. This study should result in a preliminary RDT&E plan for cost-effective systems considering the other space elements mentioned above. The required lifetimes of the components should also be evaluated in this study effort.

- Perform more detailed evaluations of the space tug astrionic module use for the RNS and four stage Saturn V missions. This includes using the astrionic module structure and components to function in either a prime or a backup role.

- Continue the system integration effort to combine subsystem components into an optimized astrionic system.

6.2 MISSION AND REQUIREMENTS ANALYSIS

The mission plan and preliminary functional requirements were developed for the space tug. The following additional effort is required to supplement the initial effort.

- Interface with other space tug contractors to update the present mission plan with the mission operational aspects defined by other space tug systems and modules. This includes how the space tugs will be launched, configured, activated, fueled, etc., in space.

- Investigate the lifetime capabilities of the space tug in a 100 nm circular storage orbit to determine the feasibility of this phase for the synchronous orbit mission. Preliminary data indicates potential problems in maintaining this orbit for long periods of time.

- Update the functional requirements for all space tug design missions and confirm the requirements for commonality. The requirements from other tug contractors will be used to aid in delta V versus N, G and C accuracy comparisons. Rendezvous schemes versus mission times and delta V penalties should be assessed.

6.3 DATA MANAGEMENT SUBSYSTEM EFFORT

A data management subsystem has been defined using a data bus approach for the subsystem. Additional effort is required to:

- Perform more detailed trades for the data bus concept, a centralized input/output concept or some combination of these concepts for the optimum data management configuration. This effort will determine the basis for updating data management modularity concepts.

- Continue the investigation of data management software concepts, specifically concerning the trades between the modular and conglomerate software concepts.

- Assess the data management requirements for NERVA engine control. The NERVA control appears to be a complex requirement and may have a large impact on data management requirements for the RNS mission.

- Determine the computer requirements for monitoring and controlling the automatic refueling of the space tug in space (based on the operational aspect of refueling to be determined by mission analysis).

- Perform further analysis of trades for magnetic memories, which retain their current state when power is removed, and monolithic memories which are volatile or lose their state when power is removed, but have the lowest power, weight and volume characteristics. Organization of the memories and error detection and correction should also be analyzed.

6.4 NAVIGATION, GUIDANCE AND CONTROL SUBSYSTEM EFFORT

The navigation sensors, guidance schemes and control schemes have been defined during this study. The future effort required to better define this subsystem is as follows:

- Perform a more detailed analysis in the selection of navigation sensors, comparing the components with the updated functional requirements. Specifically, the tradeoffs between the strapdown IMU and the gimballed IMU accuracies will be more extensively defined.

- Investigate the schemes used for space tug functions (such as rendezvous, docking, plane changes, etc.) to determine optimum schemes in conjunction with delta V and time penalties. (This would be in conjunction with the mission and requirements analysis.) The results of this study would then be compared with the guidance and control schemes recommended to assure compatibility.

- Perform a more detailed control analysis as the RCS and main propulsion engine characteristics are defined by other space tug contractors. The detailed analysis for the control of this equipment would also be performed.

6.5 ELECTRICAL POWER SUBSYSTEM EFFORT

The electrical power study to date has defined the electrical power requirements for the astrionic module and has made a "quick look" assessment of a centralized power subsystem for the entire tug. Additional study effort required is as follows:

- Integrate inputs from other space tug contractors to determine the impact of a centralized electrical power subsystem for the electrical power requirements of all space tug modules. An integrated power subsystem would then be sized.

- Investigate the feasibility and applicability of using an automatically managed power subsystem. The study would determine the extent of onboard data management capabilities required to effectively manage the space tug power subsystem.

- Determine the power requirements and associated power subsystem for the storage phases of the space tug missions. This would be based on equipment usage during the storage phases. The storage power requirements were not addressed in this study.

6.6 ELECTRICAL NETWORKS SUBSYSTEM EFFORT

This subsystem effort to date has estimated the power and signal distribution components for interfacing with the data management and electrical power subsystems. Future effort for this subsystem should concentrate on a better definition of electrical networks and distribution schemes.

6.7 COMMAND AND CONTROL SUBSYSTEM EFFORT

The command and control subsystem study to date has defined VHF and USB communication equipment for use on the space tug. Recommended future study effort is as follows:

- Investigate the interface requirements between the space tug and the DRSS for communications with the ground and other space elements. Specifically, the requirements for and feasibility of using the KU-band (13 to 15 GHz) for communications should be addressed.

- Determine the communication links currently being defined for space-to-space communications between the space tug and the space station, RNS or space shuttle. VHF is the choice at this time, but consideration should be given to using the USB equipment for this purpose. Consideration should also be given to the use of a VHF uplink during the storage phase.

- Investigate the use of VHF ranging (range and range rate) to supplement the laser radar equipment. The VHF equipment would be used for ranges up to 350nm. Current planning indicates that VHF ranging may be used on the space shuttle and space station.

6.8 THERMAL CONTROL SUBSYSTEM EFFORT

The thermal control study to date has defined the equipment to thermally control the astrionic equipment. Emphasis was placed on the earth orbit missions. Future study effort for the thermal control subsystem is as follows:

- Perform a more detailed analysis of the thermal control requirements for the astrionic module, assuming an integrated electrical power subsystem. Effort should be specifically oriented at determining the thermal protection required for the lunar type missions, such as lunar landing and the requirements for a protective heating shroud during launch for the four stage Saturn V mission.

- Perform trade studies to determine the feasibility of using an integrated thermal control subsystem to provide thermal protection for the propulsion and crew modules as well as the astrionic module.

6.9 INSTRUMENTATION SUBSYSTEM EFFORT

The instrumentation subsystem has not been specifically addressed during this study. An effort should be expended to determine the impact of instrumentation (sensors) on the design of the astrionic system.

6.10 STRUCTURES AND PACKAGING EFFORT

The structures and packaging study effort to date has defined preliminary structure and packaging concepts and an equipment layout for an astrionic module. Recommended future study effort for this area is as follows:

- Continue the present study effort to better define the structures and packaging concepts. This includes structure and packaging trades for maintainability, accessibility, etc. Specifically, the structure trades will determine the optimum structural concept(s) to meet the space tug modularity concept.

- Perform detailed analysis to better define the packaging concepts, especially for quick mounting and disconnect techniques for components, pipes, electrical wiring, etc.

- Identify the module-to-module interfaces required by other space tug contractors to better define the required physical and electrical interfaces between the space tug astrionic module and other space tug modules.

6.11 ONBOARD CHECKOUT EFFORT

The checkout study effort to date has defined preliminary concepts for checkout of the space tug equipment. The future study effort should further detail the trade information between the centralized control and built-in-test-equipment concepts based on the presently defined hardware for the tug. The test and software impacts should also be further defined as an input to the data management subsystem

6.12 RELIABILITY EFFORT

The reliability study to date has determined reliability schemes which may be used for reliability enhancement and the capability of the proposed astrionic equipment to meet preliminary reliability goals. Study results indicate the need for more sophisticated reliability enhancement techniques for the longer duration missions. Future study effort should:

- Assess the feasibility of changes to mission operation requirements which impose severe reliability requirements on astrionic equipment. For example, assess the reliability impact of allowing limited astrionic maintenance during the extended quiescent stay on the lunar surface.

- Reevaluate reliability goals and perform trade studies which relate astrionic design impact to reliability goals.

- Investigate the feasibility of implementing more sophisticated reliability enhancement techniques.

6.13 DISPLAY EFFORT

The display study effort to date has concentrated on multipurpose electronic displays to meet space tug display requirements. Recommended future study effort is as follows:

- Investigate other display equipment, such as the deformographic storage display tube, light emitting diodes, fiber optics and CRT to optimize the design of the space tug displays.

- Determine the display requirements and constraints imposed by other space tug contractors as an input to the design of the displays for the space tug.

APPENDIX A

SPACE TUG MISSION ANALYSIS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

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1.0 INTRODUCTION

1.1 PURPOSE

The purpose of this document is to describe the space tug design missions that determine the astrionic system design requirements and to define the groundrules and assumptions used for the conduct of the study. In addition, mission timelines, profiles, vehicle configurations and communication interface requirements are also presented.

1.2 SCOPE

Design missions for space tug are categorized as follows:

1. Low earth orbit to synchronous orbit
2. Earth orbit operations
3. Unmanned planetary
4. Reusable nuclear shuttle
5. Lunar orbit operations (similar to earth orbital operations)
6. Lunar landing
7. Four stage Saturn V

In order to arrive at the above categorization, all possible tug missions were evaluated from an operational requirements standpoint and grouped accordingly. Next the astrionic requirements for the missions within each category were analyzed and the missions which imposed the most stringent requirements were selected for detailed analysis. Therefore, the resultant design missions, which are described in this appendix, collectively impose the astrionic requirements for all space tug missions.

All mission descriptions contained in this appendix start at the beginning of the individual mission. For example, the earth orbit and unmanned planetary missions begin with the tug docked to the space station. The tugs required to initialize these various missions are assumed to have been brought to the space station using one of the options shown below.

1.3 GENERAL MISSION

The space tugs will be injected into low earth orbit (100 NM to 270 NM), either completely or partially deactivated (except for the four stage Saturn V mission), by a space shuttle or Saturn derivative vehicle. When the boost vehicle has achieved its final earth orbit altitude, there are two options for getting the space tug to the space station.

1. A space tug, previously brought to the space station, can be activated to retrieve the deactivated "new" space tug and take it to the space station for activation and fueling. The "old" tug would use the earth orbital mission profile to accomplish the mission.

2. The space tug can be activated and checked out. After becoming operational, the tug can perform the maneuvers necessary to undock from its boost vehicle and rendezvous and dock with the space station. The tug would be deactivated until required for a future mission.

2.0 GROUNDRULES AND ASSUMPTIONS

These groundrules and assumptions form the background for this study. The baseline space tug vehicle configuration used for this study is shown in Figure 2-1.

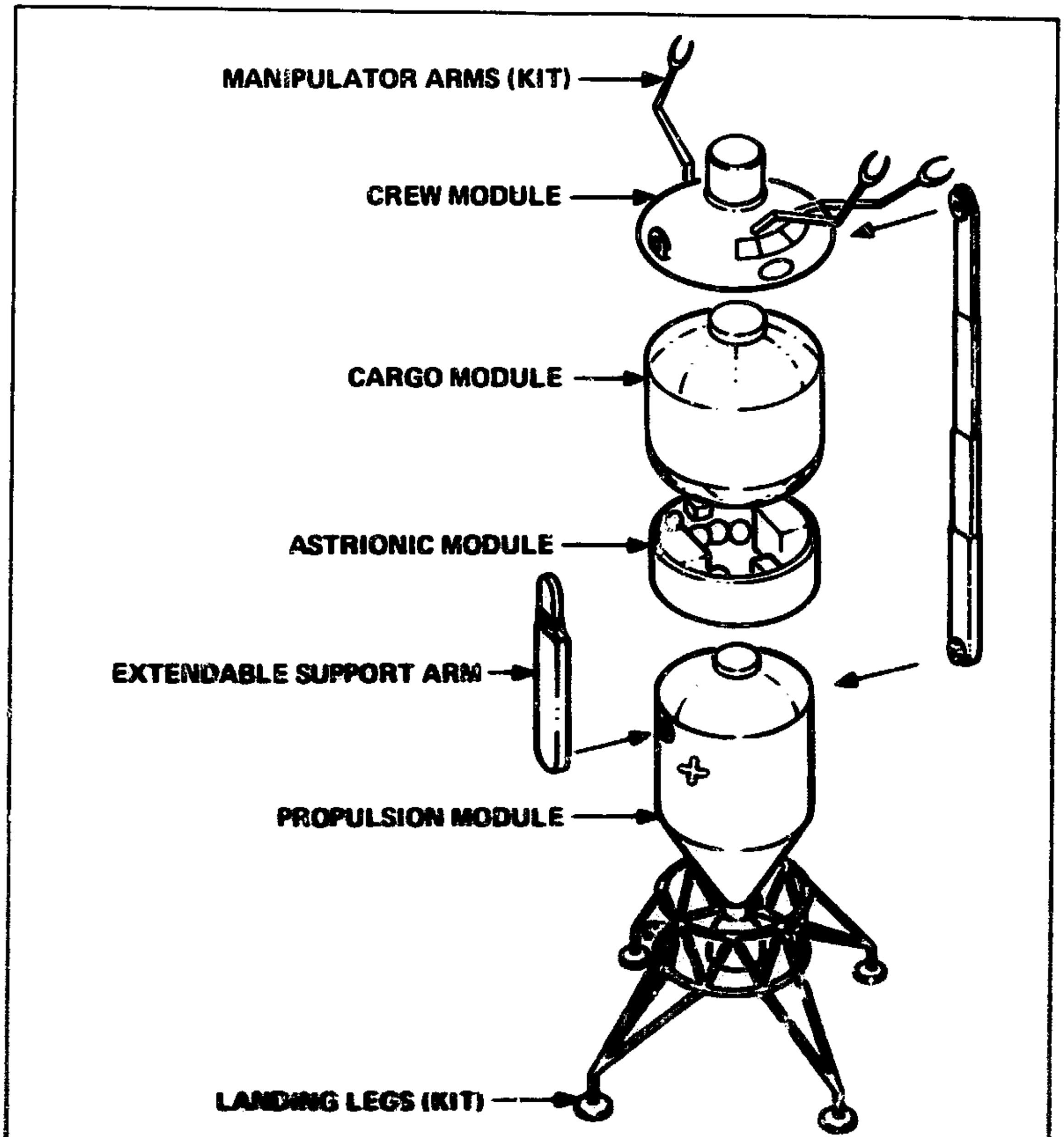

Figure 2-1 Space Tug Baseline Configuration

- The space tug includes four basic modules (crew, propulsion, astrionic, and cargo) and auxiliary kits for special-purpose missions (landing legs, manipulation arms, etc.) as required. Combinations of the modules and kits will be used for each mission.
- The tug is based and maintained in space and on the ground.
- The tug may be configured for manned or unmanned missions. In the unmanned configuration, the tug will be operated automatically or by remote control from the earth or from other space elements.
- The space tug will be delivered to orbit by a space shuttle or Saturn derivative vehicle.
- The lunar orbit space station is assumed to be in a polar orbit at a circular altitude of 60 nautical miles.
- The space tug shall be compatible with the earth and lunar orbiting space stations, the space shuttle, and the reusable nuclear shuttle (RNS).
- The space tug astrionic module design shall minimize the need for ground support.
- The space tug shall be capable of maintaining a quiescent status for up to 180 days in earth or lunar orbit when docked to other space elements or free-flying. Quiescent periods of up to 42 days will be required on lunar surface (14 or 28 days + 14 days contingency).
- The space tug shall be capable of going from the quiescent state to a fully operational state within two hours.
- The reusable space tug shall have a minimum lifetime goal of ten years and the capability of being reused at least ten times by replenishment of consumables and through performance of required maintenance. Maintenance, as required, and reconfiguration capability while in space residence is considered mandatory. Major refurbishment may necessitate the need for the tug to be returned to earth.
- Maximum crew safety and a high probability of fulfilling all space tug functions and objectives shall be a design goal. Subsystems identified as necessary for crew survival will be designed such that no single failure or credible combination of failures will result in loss of life.
- In the manned mode, the space tug can be piloted by one crewman.
- The crew module will serve as the primary crew living quarters and a base of operations (mission control) for manned missions.
- The crew module will contain an airlock for EVA and multi-simultaneous EVA operation capabilities.

- Tug communication systems will be compatible with the Manned Space Flight Network, Deep Space Network, SGLS (DOD), available Communication Satellite Systems and with space elements, such as the space stations, shuttles, etc., depending on the mission.

- The space tug attitude reference system shall have complete freedom in all axes.

- The space tug shall have neuter docking devices compatible with all space vehicle hardware elements.

- Minimum interfaces are required between the space tug and its payload to reduce complexity and increase the flexibility of the kinds of payloads to be transported. However, consideration should be given to how the space tug communications and power subsystems could support the payload.

- The primary propulsion system will be LOX/LH_2. Secondary propulsion systems may also be LOX/LH_2.

- The first operational flight will be flown no earlier than 1977.

- The tug astrionic system will have the capability to automatically control and monitor the refueling process with automatic shutoff and disconnect capability.

- The synchronous orbit mission of the tug will be a baseline.

- Only cooperative satellites are considered for all retrieval missions.

- More than one low-energy mission may be performed between refuelings.

- The space tug maximum diameter must be less than 15 feet. One throttleable and gimballed engine will be used.

- The space tug astrionic system shall not be constrained by lunar lighting conditions. The tug shall be capable of landing at all lunar latitudes and longitudes with appropriate time-phasing.

DEFINITIONS

- Remote operation The control of the system is accomplished from an external source.

- Automatic operation The system is completely independent of any external control or astronauts

- Cooperative satellites Satellites that are attitude stabilized, have corner reflectors, and have a docking adapter(s)

3.0 SYNCHRONOUS ORBIT MISSION (REUSABLE)

3.1 GUIDELINES

Vehicle: Two unmanned reusable space tugs (propulsion module, astrionic module), one set of manipulator arms, and one payload (cargo module).

Payload: Synchronous orbit satellite.

Mission: Provide the propulsion vehicle to place a payload into synchronous equatorial orbit (with a 28.5^{o} plane change) and return the payload to earth orbit (with a 28.5^{o} plane change). The mission will begin in a 100 NM circular orbit at an inclination of 28.5^{o}.

Operation: Unmanned. Unmanned vehicles will have the capability of automatic or remote operations. Possibility of a manned mission to synchronous orbit does exist.

Frequency of Operation: Assume reusable tugs will be used continuously for a maximum of 60 hours.

3.2 MISSION DESCRIPTION

The initial tug operation will begin with the space tug in the cargo bay of the space shuttle and the space shuttle ready for launch. The shuttle will then transport the payload (tug) to a 100 NM circular orbit at an inclination of 28.5^{o}.

When the space shuttle has reached orbit, the space tug will be fully activated. It will be jettisoned from the shuttle cargo bay and will remain in this orbit prior to final assembly for the mission. The second reusable space tug will be brought into orbit and assembled with the first tug. At this point the baseline mission will begin.

The space tugs will be activated and checked out. After undocking from the shuttle, the tugs will maneuver from the shuttle. When they are properly phased for the synchronous orbit target, a Hohmann transfer burn will be used to transfer the payload to synchronous orbit. The first tug will burn to start the payload on its journey, will separate and prepare for a deorbit maneuver. The second tug will ignite immediately after separation and provide the additional impulse to place the payload in an approximately 19,300 NM by 100 NM orbit.

The first tug will perform a second burn to deorbit and a third burn to circularize at a 100 NM orbit. The tug will then rendezvous and dock with the shuttle and, after docking, will be deactivated.

The second tug will coast approximately one-half revolution after its initial burn and perform a second burn at apogee to change the plane of the orbit to equatorial inclination and circularize at approximately 19,300 NM. Vernier burns and maneuvers will be used for the final rendezvous to the target. The tug will then place the payload and maneuver to and dock with another satellite that is in synchronous orbit. The space tug will be in a station keeping status during this period.

After the payload transfer is complete, a retrograde burn will lower the orbit to 19,300 NM by 100 NM and accomplish a plane change of 28.5^{o}. One-half revolution later at perigee, a second transfer burn will circularize the orbit at approximately 100 NM. The tug will then rendezvous and dock with the first tug at the space shuttle. The space shuttle will either (1) bring the tug(s) back to earth for refurbishment along with the payload or (2) return the payload and the tugs will be placed in a quiescent mode and stored in earth orbit for a maximum of 180 days and until refueled by another space shuttle mission. Both the first and second space tugs will remain in orbit until required for another mission.

The mission profile and space tug vehicle configuration for this mission are shown in Figure 3-1.

3.3 DETAILED MISSION PHASE DESCRIPTIONS

The space tug is assumed to be launched into a 100 NM circular orbit at an inclination of 28.5^{o} in the cargo bay of the space shuttle. Space tug active mission will begin when both space tugs are fueled and configured for the mission.

3.3.1 Activation

After both space tugs reach orbit, both astrionic systems will be completely activated and checked out using the tug onboard checkout system. The onboard equipment will be initialized with mission parameters for the specific mission. The mission parameters will be received from either the space shuttle or the ground mission control.

3.3.2 Undocking and Orbital Coast

Space tugs will undock from the space shuttle and will begin operational control. The tugs will maneuver a safe distance from the space shuttle prior to the tugs' first main burn. The tugs will coast in low earth orbit until achieving the proper phasing for the desired synchronous target.

At this point, the mission phases can be divided into Part A, Space Tug 1 Mission, and Part B, Space Tug 2 Mission.

Part A – Space Tug 1 Mission

3.3.3A Transfer Burn No. 1

When the tug is properly phased for the synchronous target, the tug will perform the initial part of the Hohmann transfer burn to achieve an elliptical orbit of approximately 19,300 NM by 100 NM.

3.3.4A Separation

After completion of the transfer burn the lower space tug will separate and maneuver to an attitude for the deorbit burn.

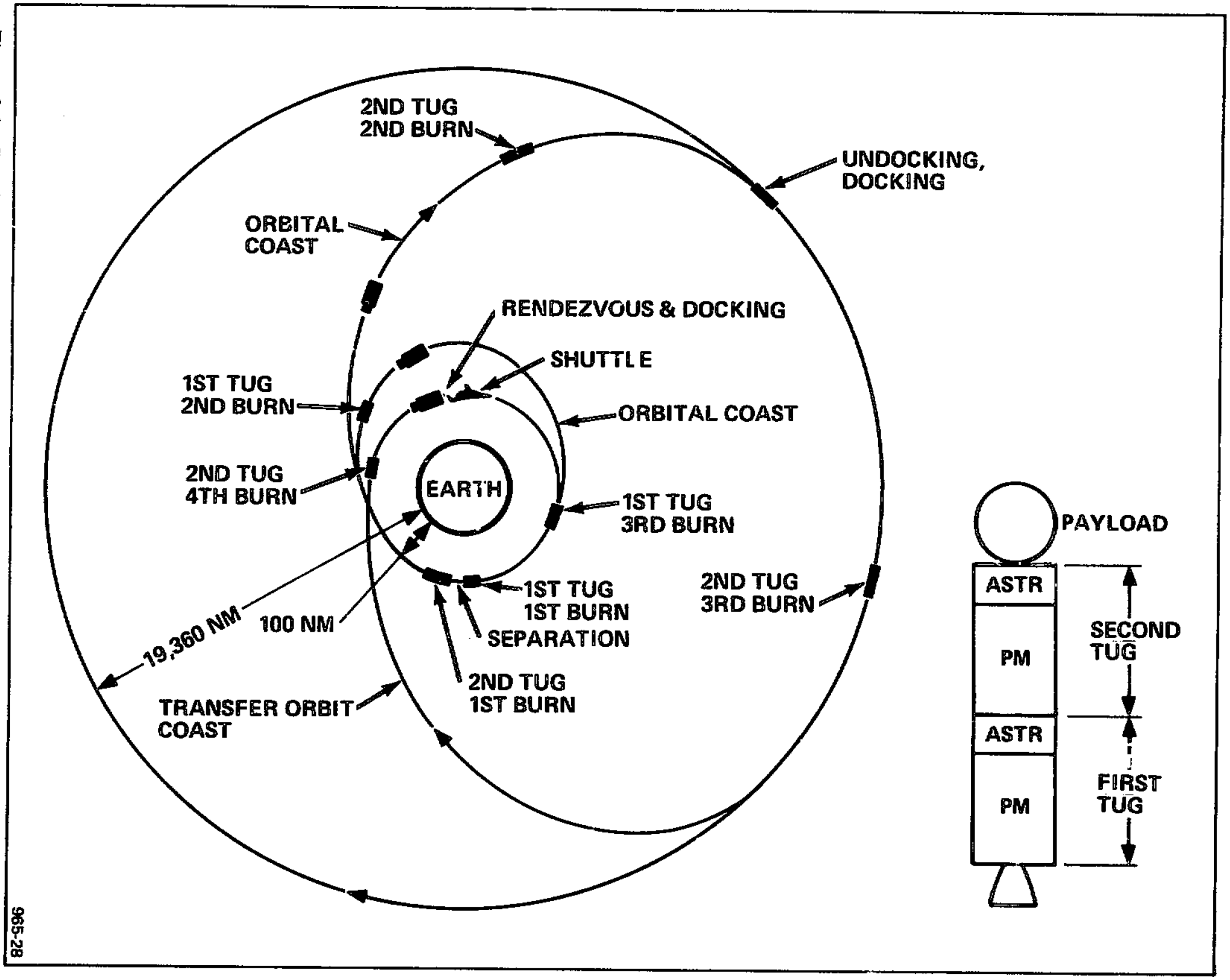

Figure 3-1. Synchronous Orbit Mission Profile and Vehicle Configuration
(Reusable Tugs)

3.3.5A Orbital Coast No. 1

The tug will coast during the attitude maneuver until it is a safe distance from the second tug.

3.3.6A Transfer Burn No. 2

The transfer burn will allow the space tug to burn to achieve an orbit whose perigee is 100 NM.

3.3.7A Orbital Coast No. 2

Space tug will coast in the elliptical orbit until achieving a perigee of 100 NM.

3.3.8A Transfer Burn No. 3

This transfer burn will allow the circularization of the tug into a 100 NM circular orbit.

3.3.9A Rendezvous and Docking

The combination of short duration burns in coast periods will be used to rendezvous the tug with the space shuttle. When the tug has completed rendezvous with the space shuttle, it will dock with the space shuttle.

3.3.10A Deactivation

When the tug is attached to the shuttle, the tug will be deactivated. The tug will either be brought back to earth, where it will be refurbished, or left in earth orbit where it will be deactivated and stored for a period not to exceed 180 days.

Part B – Space Tug No. 2 Mission

3.3.3B Transfer Burn No. 1

As soon as the tug is separated from the other space tug, a burn will be accomplished to add the additional thrust necessary to complete the Hohmann transfer and achieve an elliptical orbit of approximately 19,300 NM by 100 NM.

3.3.4B Orbital Coast No. 1

The tug will coast for approximately one-half revolution or until the tug reaches apogee (19,300 NM). Any required target or navigation update will be received during this period.

3.3.5B Transfer Burn No. 2

At apogee, a second transfer burn for the tug will be used to circularize the orbit at approximately 19,300 NM. This burn will be a combination maneuver to circularize the orbit and accomplish a plane change to place the payload in an earth equatorial synchronous orbit.

3.3.6B Orbital Coast No. 2

The tug will coast in preparation for final rendezvous operations. Any target or navigation update will be received during this phase.

3.3.7B Rendezvous

A combination of vernier burns and coast periods will be used to maneuver the tug and place the payload in an exact equatorial synchronous orbit of 19,363 NM.

3.3.8B Station Keeping

The space tug will maintain attitude control in a synchronous orbit while the payload is undocked. After the payload is operational, the space tug will maneuver to another previously placed synchronous satellite and dock to that satellite for return to lower earth orbit. The tug will maintain an active status and attitude control for the duration of this phase.

3.3.9B Transfer Burn No. 3

After the mission has been completed the tug will perform a retrograde burn to deorbit the tug. This burn will perform the 28.5° plane change maneuver and place the tug in an elliptical 19,363 NM by 100 NM orbit.

3.3.10B Orbital Coast No. 3

The tug will coast for one-half revolution or until perigee (100 NM). Any target or navigation update will be received during this coast period.

3.3.11B Transfer Burn No. 4

The final main burn of the tug will be a retrograde burn at perigee to place the tug in an approximately 100 NM circular orbit at an inclination of 28.5°.

3.3.12B Rendezvous and Dock with Space Shuttle

The combination of short duration burns and coast periods will be used to rendezvous the tug with the space shuttle. When the tug has completed rendezvous with the shuttle, it will dock with the first tug.

3.3.13B Deactivation

When the tugs are docked, they will be activated and either brought back to earth by the shuttle where they will be refurbished or left in earth orbit where they will be deactivated and stored for a maximum of 180 days. If stored, the onboard systems required to insure the integrity of the astrionics may be active during this phase.

3.4 INTERFACE REQUIREMENTS

In order to allow remote control of the tug and manipulator, constant communications is required with the ground and/or space shuttle when docking or undocking with the payload. The communications consist of TV, telemetry, command and tracking.

The communication interface should be a USB type system for simplicity of ground interface. The ground interface for NASA missions should be to mission control, probably via Goldstone, Madrid, and/or Honeysuckle.

The ground interface for DOD missions should be the Satellite Control Facility via a DOD satellite system or the network controlled by the Satellite Control Facility. In both cases, the interface should be a USB type (USB/NASA, SGLS/DOD).

3.5 MISSION TIMELINE

Initial Conditions

2 Tugs mated, fueled, payload is attached, and configuration is docked with Space Shuttle.

	DURATION	SPACE TUG NUMBER 1 MAX CUMULATIVE TIME	SPACE TUG NUMBER 2 MAX CUMULATIVE TIME
BOTH:			
Activation	1-2 hours	2.0 hours	2.0 hours
Undock (Tug from Space Shuttle) and Orbital Coast	1.5 hours	3.5 hours	3.5 hours
SPACE TUG NUMBER 1			
Transfer Burn Number 1	4-8 minutes	3.6 hours	3.6 hours
Separation	–		
Orbital Coast Number 1	0.5-2 hours	5.6 hours	
Transfer Burn Number 2	4-8 minutes	5.8 hours	
Orbital Coast Number 2	2-4 hours	9.8 hours	
Transfer Burn Number 3 (100 NM Circular)	1-4 minutes	9.9 hours	
Rendezvous and Dock		15.9 hours	
with Space Shuttle	2-6 hours	17.9 hours	
Deactivate	1-2 hours	17.9 hours max.	
SPACE TUG NUMBER 2			
Transfer Burn Number 1	1-4 minutes		3.7 hours
Orbital Coast Number 1	5-6 hours		9.7 hours
Transfer Burn Number 2 (Circularization and plane change)	1-4 minutes		9.8 hours

(Continued on next page)

	DURATION	SPACE TUG NUMBER 1 MAX CUMULATIVE TIME	SPACE TUG NUMBER 2 MAX CUMULATIVE TIME
SPACE TUG NUMBER 2			
Orbital Coast Number 2	1-4 hours		13.8 hours
Rendezvous	2-6 hours		19.8 hours
Station Keeping	2-24 hours		43.8 hours
Transfer Burn Number 3	4-8 minutes		44.9 hours
Orbital Coast Number 3	5-6 hours		50.9 hours
Transfer Burn Number 4 (100 NM Circular)	1-4 minutes		51.0 hours
Rendezvous and Dock with Space Shuttle	2-8 hours		58.0 hours
Deactivate	1-2 hours		60 hours max.
Storage	30-180 days	180 days max.	180 days max.

4.0 SYNCHRONOUS ORBIT (EXPENDABLE)

4.1 GUIDELINES

Vehicle: One unmanned expendable space tug (propulsion module and astrionic module)

Payload: Synchronous orbit satellite

Mission: Provide the propulsion vehicle to place a payload in synchronous equatorial orbit (with a 28.5° plane change). This mission will begin in a 100 NM circular orbit at an inclination of 28.5°. The payload and space tug will be brought up to the 100 NM circular orbit using the space shuttle.

Operation: Unmanned with the capability of automatic or remote operations.

4.2 MISSION DESCRIPTION

The initial mission will begin with the space tug and payload in the cargo bay of the space shuttle at a 100 NM circular orbit with an inclination of 28.5°.

The space tug will start the mission docked to the space shuttle. The tug will be activated and checked out and will undock from the shuttle. When the tug is properly phased for a synchronous orbit target, a Hohmann transfer burn will place the tug in an approximately 19,300 NM by 100 NM orbit. At apogee, a second burn will be accomplished with a plane change of 28.5° to put it in an equatorial orbit and circularize this orbit at approximately 19,300 NM. Vernier burns and maneuvers will be used for the final rendezvous for proper placement of the payload.

After the mission is complete, the expendable space tug will be placed in an attitude for burning out of orbit and will be burned to depletion, thereby disposing of the tug and placing it in some orbit outside the synchronous altitude. The mission profile and space tug vehicle configuration for this mission are shown in Figure 4-1.

4.3 DETAILED MISSION PHASE DESCRIPTIONS

The space tug is assumed to be launched inactive into a 100 NM circular orbit at an inclination of 28.5° in the cargo bay of the space shuttle.

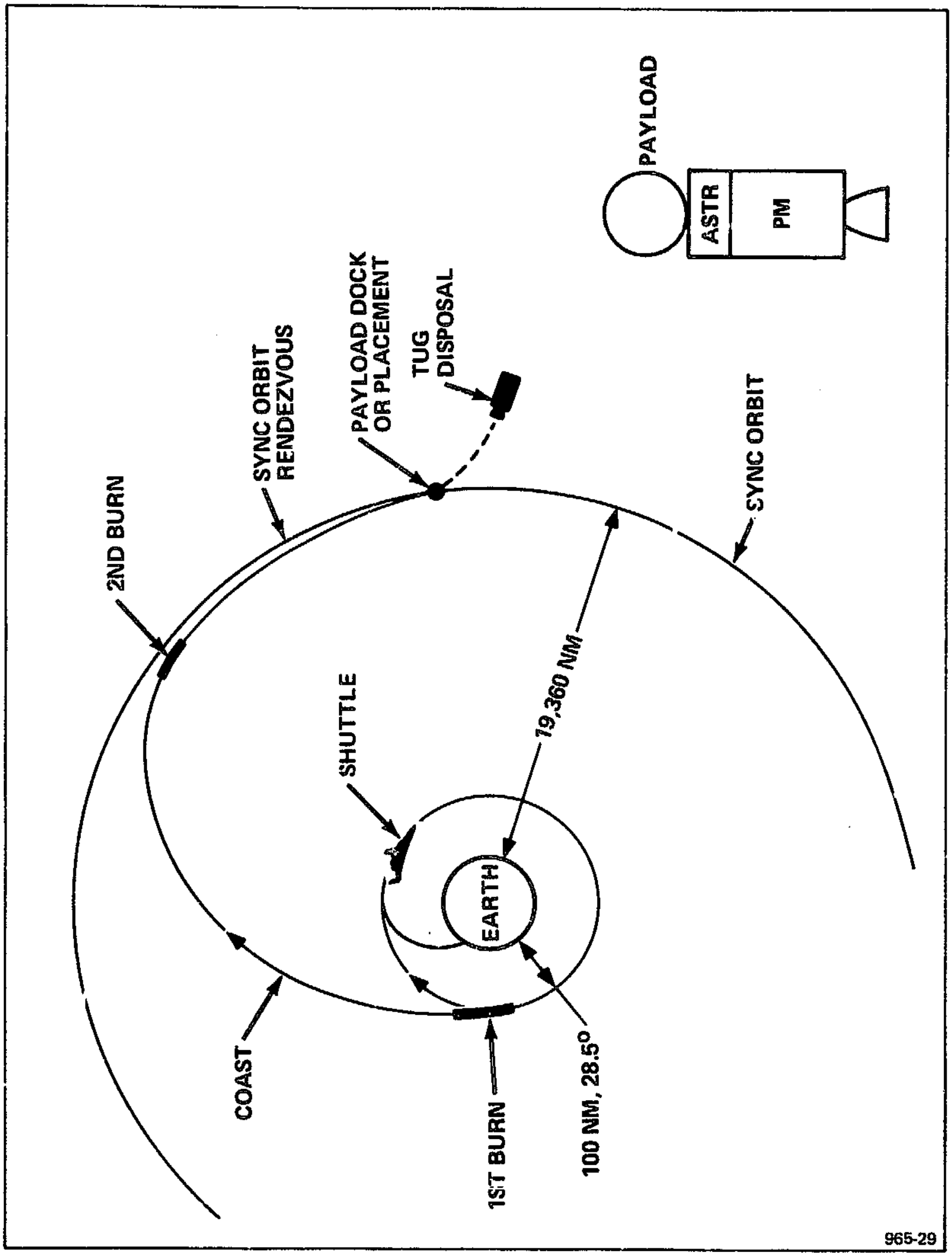

Figure 4-1. Synchronous Orbit Mission Profile and Vehicle Configuration (Expandable Tug)

4.3.1 Activation

After the space tug reaches orbit, the astrionic system will be completely activated and checked out using the tug onboard checkout system. The onboard equipment will be initialized with mission parameters for the specific mission. The mission parameters will be received from either the space shuttle or the ground network.

4.3.2 Undocking

The space tug will undock from the cargo hold of the space shuttle and will begin operational control. The tug will maneuver a safe distance from the space shuttle prior to its first main burn.

4.3.3 Transfer Burn No. 1

When the space tug is properly phased for the synchronous target, the space tug will perform the initial part of the Hohmann transfer burn to achieve an elliptical orbit of approximately 19,300 NM by 100 NM.

4.3.4 Orbital Coast No. 1

The space tug will coast for approximately one-half a revolution or until the space tug reaches apogee (19,300 NM). Any required target or navigation update will be received during this period from the ground.

4.3.5 Transfer Burn No. 2

At apogee, the transfer burn for the space tug will be used to circularize the orbit at approximately 19,300 NM. This burn will be a combination maneuver to circularize the orbit and accomplish a plane change (28.5°) to place the payload in the earth equatorial synchronous orbit.

4.3.6 Orbital Coast No. 2

The space tug will coast in preparation for final rendezvous operations. Any target or navigation update will be received during this phase.

4.3.7 Station Keeping

The space tug will maintain attitude control in synchronous orbit while the payload is undocked. After the payload is operational, the space tug will maneuver to an attitude not to interfere with any other payloads in synchronous orbit and burn to depletion.

4.3.8 Transfer Burn No. 3

This burn will be a combination of main propulsion and attitude control. The burn will be accomplished after achieving the out-of-plane maneuver to allow for depletion of all propellants on board and also to allow for disposal of the space tug, preventing it from interfering with other objects in synchronous orbit.

4.4 INTERFACE REQUIREMENTS

In order to allow remote control of the space tug, constant communication is required with the ground and/or space element when docking with or undocking from the payload. The communications will consist of TV, telemetry, and command. Target updating may also be required; therefore, tracking must be available.

The communications to support this type of interface should be a USB type of system for simplicity of ground interface. Since the space tug will not be in constant communication with a space element, the ground interface should be utilized. The ground interface for NASA missions will probably be mission control via Goldstone, Madrid, and/or Honeysuckle, since all three stations will have approximately 270-foot antennas. The DRSS may be used when the space tug is in a low earth orbit.

The ground interface for DOD missions should be the Satellite Control Facility via either a DOD satellite system or the network controlled by the Satellite Control Facility. In both cases, the interface should be a USB type (USB/NASA, SGLS/DOD).

4.5 MISSION TIMELINE

PHASE	DURATION	MAXIMUM ACCUMULATIVE TIME
Activation	1 – 2 hours	2 hours
Undocking	0.5 hours	2.5 hours
Transfer Burn No. 1	4 – 8 minutes	2.6 hours
Orbital Coast No. 1	5 – 6 hours	8.6 hours
Transfer Burn No. 2	1 – 4 minutes	8.7 hours
Orbital Coast No. 2	1 – 4 hours	12.7 hours
Station Keeping	2 – 6 hours	18.7 hours
Transfer Burn No. 3	To Depletion	18.9 hours max.

5.0 ORBITAL OPERATIONS MISSION

5.1 GUIDELINES

Vehicle: One reusable space tug (propulsion module, astrionic module, crew module, payload).

Mission: Space resident vehicle used for payload/passenger transfer in conjunction with a space station in earth or lunar orbit.

Similar Additional Missions:

(a) Rescue vehicle for earth or lunar orbit

(b) Satellite repair or transfer

Launch Vehicle: Space shuttle, INT-21 or another Saturn derivative. Space tug inactive during launch.

Storage: Up to 180 days of quiescent operation, either docked or free-flying, at approximately 270 NM altitude. Operational within two hours.

Operation: Manned or unmanned. Capable of being remotely operated.

Frequency of Operation: Assume at least one mission every two weeks with less than 24 hours continuous operation per use.

5.2 MISSION DESCRIPTION

The space tug will be launched into low earth orbit aboard a space shuttle or a Saturn derivative vehicle. The space tug will be inactive during the launch to orbit. After the launch vehicle has completed its orbital maneuvers, the space tug may be activated and will rendezvous and dock with the Space Station at approximately a 270 NM altitude with an inclination of 28^o or 55^o. The space tug could also be delivered inactive to the space station by another space tug.

The space tug mission will begin with the space tug docked to the space station and in a quiescent mode. To begin a mission, the space tug is activated and its onboard subsystems checked out to assure operational capability. The onboard systems would then be initialized with mission parameters. The space tug would undock from the docking port of the space station and begin maneuvers to rendezvous with the payload at 100 NM for orbital transfer of a payload to the space station. Using a combination of transfer burns, the tug will maneuver to the vicinity of a space vehicle (shuttle, Saturn derivative vehicle) and dock to the payload. The space tug will then deliver the payload to the space station. After delivering the payload, the tug will either be docked to the space station or left in a free-flying mode during the storage period prior to another active mission.

5.3 DETAILED MISSION PHASE DESCRIPTIONS

The space tug will be launched into low earth orbit aboard a space shuttle or a Saturn derivative vehicle and transferred to the space station. The space tug is assumed to be docked to the space base before being activated. The following are the detailed mission phases for this mission. Figure 5-1 shows the mission profile and the vehicle configuration for the earth orbital operations mission. The lunar orbital operations missions are considered to have essentially the same requirements as the earth orbit mission, and its profile is shown in Figure 5-2.

5.3.1 Activation/Reactivation

The space tug will be activated and its astrionic system checked out to assure that all modules are operational. After completing activation, the onboard subsystems will be initialized with mission parameters.

5.3.2 Undocking

The space tug will undock from the space station and maneuver a safe distance away from the station prior to its initial burn. The space tug will be visually inspected to detect damage by a member of the space station crew as the space tug maneuvers from the station.

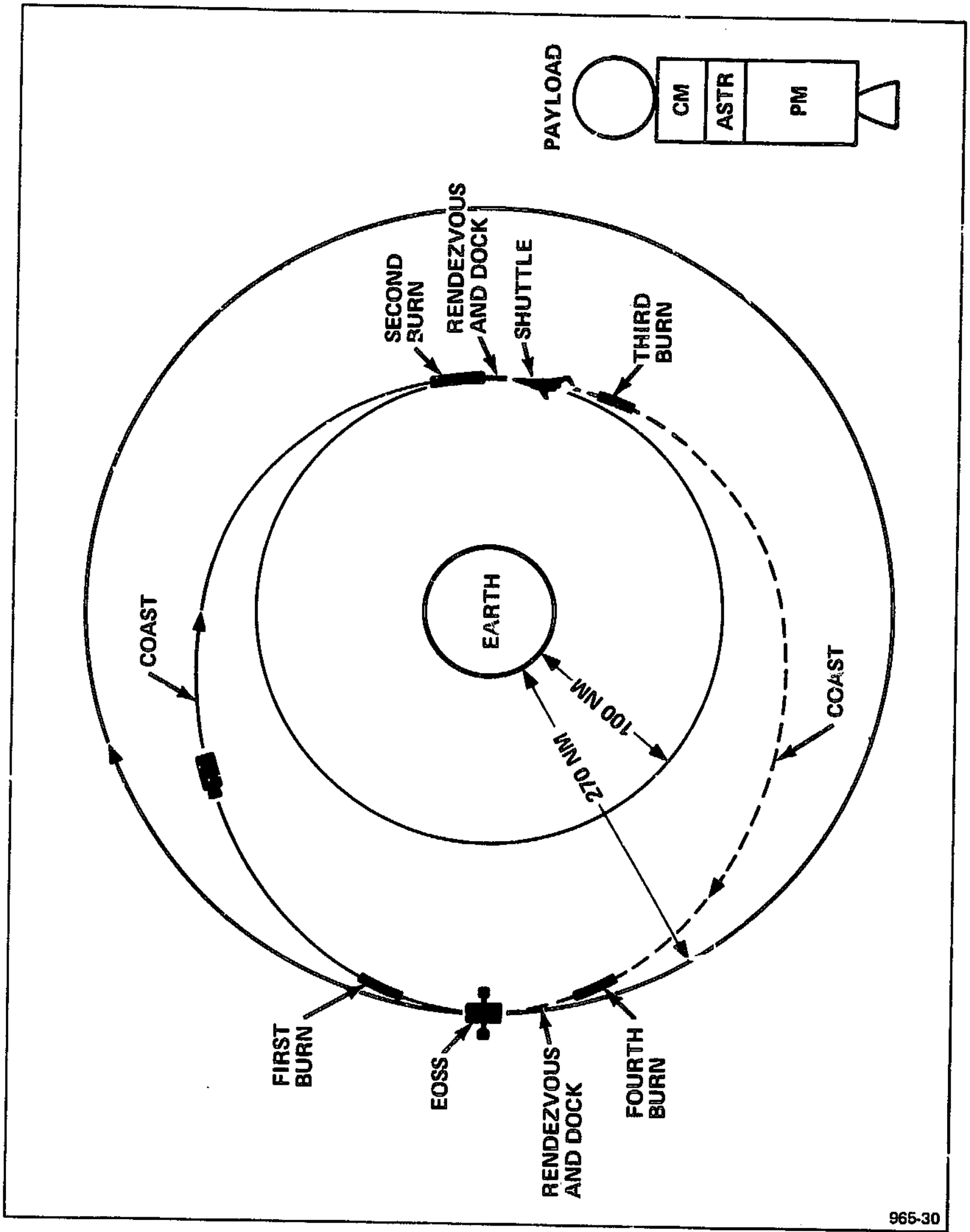

Figure 5-1. Earth Orbital Operations Mission Profile and Vehicle Configuration

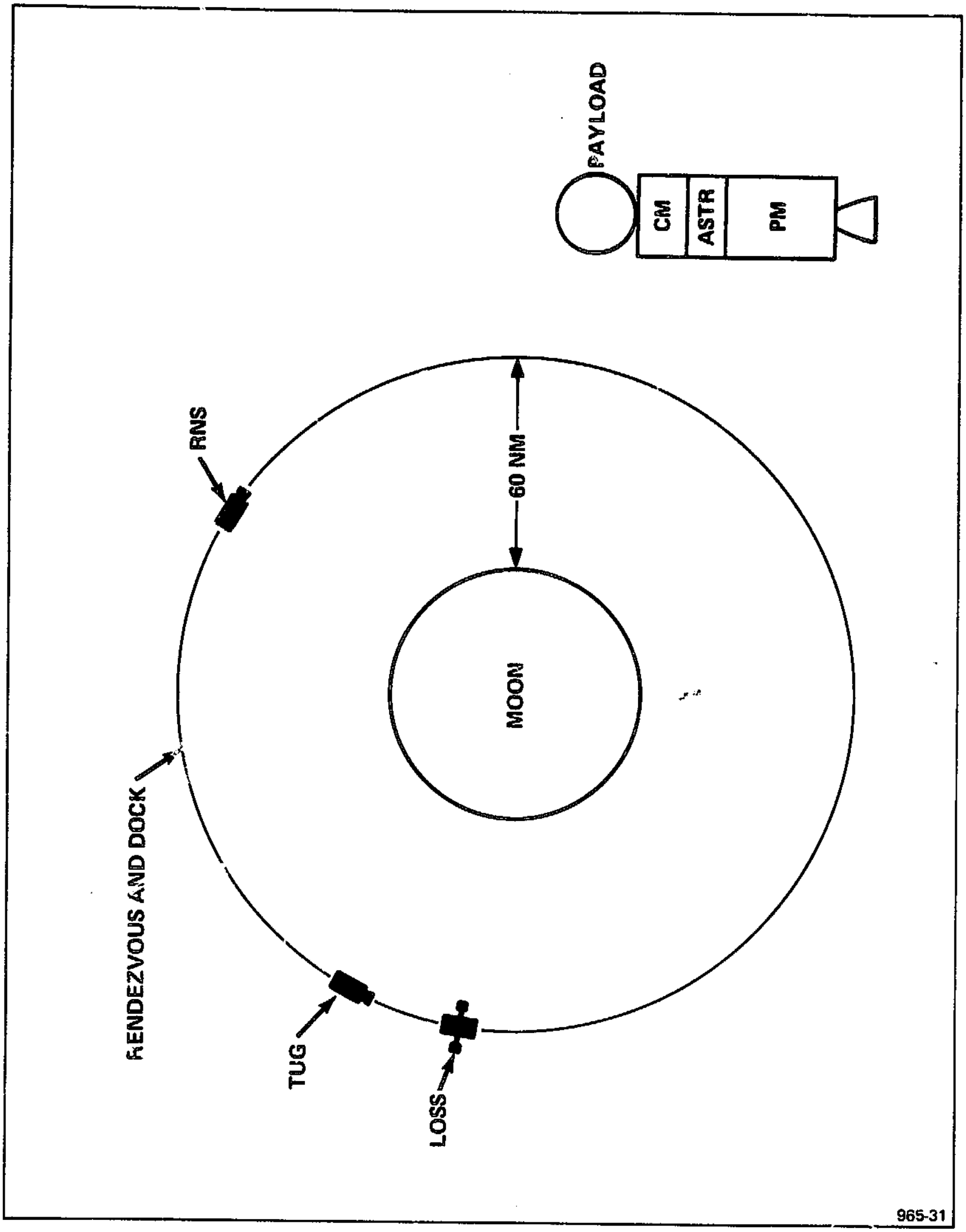

Figure 5-2. Lunar Orbit Operations Mission Profile and Vehicle Configuration

5.3.3 Transfer Burn No. 1

When the tug is a safe distance from the space station, the tug will perform a braking burn to achieve the desired transfer ellipse to place it in an approximately 270 NM by 100 NM orbit.

5.3.4 Orbital Coast No. 1

After completion of the transfer burn, the tug will coast in the transfer ellipse until time for the circularization burn.

5.3.5 Transfer Burn No. 2

On reaching the desired altitude and/or perigee, a second burn is executed to circularize the orbit at approximately a 100 NM altitude.

5.3.6 Rendezvous and Docking

After achieving the desired orbit, the space tug will rendezvous and dock to its target. The final docking will be either manual or by remote control. The space tug will remain docked to the target while the payload is transferred to the tug.

5.3.7 Undocking

After the payload has been transferred to the tug, the space tug will undock from the vehicle and maneuver a safe distance from it.

5.3.8 Transfer Burn No. 3

When the space tug is a safe distance from the space vehicle, the space tug will perform a transfer burn to achieve the desired transfer trajectory to intercept the space station. The resulting orbit will be approximately 100 NM by 270 NM.

5.3.9 Orbital Coast No. 2

After completion of the transfer burn, the tug will coast in the transfer ellipse until time for the circularization burn.

5.3.10 Transfer Burn No. 4

On reaching the space station altitude, the tug will perform a burn to place the tug in a circular orbit at approximately 270 NM.

5.3.11 Rendezvous and Docking

After completion of the circularization burn, the space tug will perform a series of short burns to rendezvous and redock to the space station.

5.3.12 Deactivation

After the tug is docked to the space station, the payload will be transferred from the tug to the space station, and the space tug will be checked out and deactivated to a quiescent status.

5.3.13 Storage

In the quiescent status, the space tug will either be docked to the space station or placed in a free-flying mode near the space station. The tug will remain in this mode until activated for maintenance, refueling or another mission.

5.3.14 Maintenance

Prior to all missions, the tug will receive a check out. During this maintenance, any short life or defective equipment will be replaced and the subsystems returned to optimum operational status. The frequency of maintenance will depend on the number of missions flown, time in space of the vehicle, past failure rates, etc. Inflight maintenance will be limited to automatic switch-in of redundant components.

5.3.15 Fueling

Prior to each mission the tug will be fueled, if necessary, with the quantity of fuel required for that mission plus contingencies. The tug will actively monitor and control the fueling process.

5.4 INTERFACE REQUIREMENTS

Due to the requirement for remote control and target updates the tug has a requirement for a communications interface with the ground, space station, and other active space vehicles.

A. Ground: A direct interface to ground (USB) or relayed through the DRSS to mission control. This interface includes command, telemetry, voice (if manned), tracking, and TV.

B. Space Station and Other Active Space Vehicles: This interface should be a VHF or USB type and include command, telemetry, voice (if manned), tracking, and TV.

The TV requirement is to allow remote control of the manipulator arms.

5.5 MISSION TIMELINE

PHASE	DURATION	MAXIMUM ACCUMULATIVE TIME
Activation/Reactivation	2.0 hours	2.0 hours
Undocking	0.5 hours	2.5 hours
Transfer Burn	0.1–1.0 minute	2.5 hours
Orbital Coast	0.8 hours	3.3 hours
Transfer Burn	1.0–2.0 minutes	3.3 hours
Rendezvous and Docking	2.0–8.0 hours	11.3 hours
Undocking	0.5 hours	11.8 hours
Transfer Burn	0.1–1.0 minute	11.8 hours
Orbital Coast	0.8 hours	12.6 hours
Transfer Burn	1.0–2.0 minutes	12.6 hours
Rendezvous and Docking	2.0–8.0 hours	20.6 hours
Deactivation	3.4 hours	24.0 hours max.
Storage	Variable	180 days max.

6.0 PLANETARY MISSION – UNMANNED

6.1 GUIDELINES

Vehicle: One reusable space tug (propulsion and astrionic module) and one expendable space tug (propulsion and astrionic module)

Payload: Planetary probe.

Mission: Provide the propulsion vehicle for planetary probes.

Operation: Unmanned, capable of automatic or remote operation.

Frequency of Operation: Expendable tugs are used once. Assume reusable tugs will be used for more than one mission.

6.2 MISSION DESCRIPTION

The mission will begin with a fueled space vehicle configured to the baseline configuration shown in Figure 6-1. The mission profile is also depicted in Figure 6-1.

The space tug astrionics will be activated, checked out to assure operational capability of all systems, and the onboard systems initialized with mission parameters. The first tug astrionics will be configured to perform the necessary maneuvers required to return to the space station after separation.

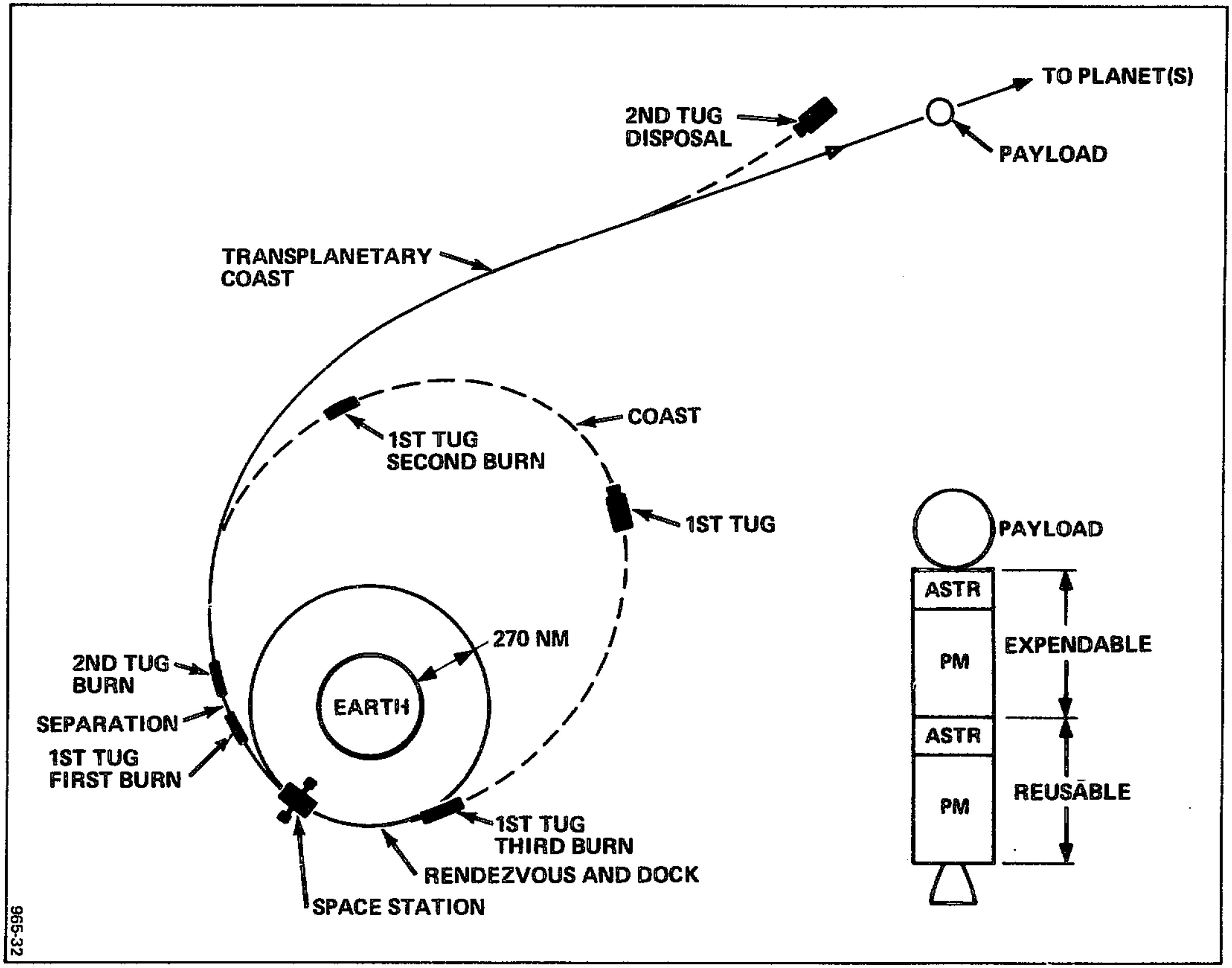

Figure 6-1. Unmanned Planetary Mission Profile and Vehicle Configuration

When the vehicle is properly phased for the burn, the first space tug will ignite to start the payload on its journey. The first space tug will burn a portion of its fuel, separate from the payload and return intact to the space station. The second space tug will ignite immediately after first tug separation. The second tug will boost the payload to the desired cutoff conditions. At this point the engine will be shut down, and the space tug will separate from the payload. It is assumed that the payload will then have the capability to perform its own navigation, guidance, control and communications functions to properly carry out its mission.

Space tug No. 1 will be deactivated and stored at the space station. After maintenance and refueling, it will be capable of performing other missions.

Space tug No. 2 will be operated in an expendable mode. After its separation from the payload, its trajectory will be altered to assure that there will be no contact with the payload. After safing of the tug, its mission will be complete.

6.3 DETAILED MISSION PHASE DESCRIPTIONS

6.3.1 Activation

The space tugs will be activated and their astrionic systems checked out to assure all modules are operational. The onboard subsystems will then be initialized with mission parameters. The second space tug will assume the operational control of the mission aided, if required, by the first tug astrionics.

6.3.2 Orbital Coast

After the checkout phase is complete, the space vehicle will be oriented to the desired attitude prior to engine ignition.

6.3.3 Transfer Burn No. 1

When the proper engine start conditions have been satisfied, the engine of space tug No. 1 will be ignited and a partial burn will be accomplished. At the appropriate time, the engine will be shut down.

6.3.4 Undocking

After engine cutoff, vehicle attitude will be maintained, and space tug No. 1 will separate from the rest of the vehicle.

At this point, the mission phases can be divided into Part A, space tug No. 1 mission and Part B, space tug No. 2 mission.

Part A – Space Tug No. 1 (Reusable)

6.3.5A Orbital Coast

After separation, the space tug will remain in the coast mode until time for the next transfer burn. Prior to the transfer burn, the tug will be oriented to the desired attitude by the RCS. Target or navigation update will be received as required.

6.3.6A Transfer Burn No. 2

At the appropriate time, the engine will be ignited to perform a retrograde burn which will place the tug in an elliptical orbit with a perigee of approximately 300 NM.

6.3.7A Orbital Coast

After engine cutoff, the space tug will remain in the coast mode until the next transfer burn. Prior to the transfer burn, the tug will be oriented to the desired attitude. Any small trajectory alterations necessary during the coast period will be accomplished. Target or navigation update will be received as required.

6.3.8A Transfer Burn No. 3

At or near perigee, the space tug engine will again be ignited to circularize its orbit at approximately 300 NM for subsequent rendezvous with the space station at 270 NM.

6.3.9A Rendezvous

Using a combination of short burns and coast periods, the space tug will rendezvous with the space station either automatically or by remote operation.

6.3.10A Dock

The space tug will perform the appropriate maneuvers, either in the automatic or remote operational mode, to accomplish docking with the space station.

6.3.11A Deactivate

The space tug will be checked out and deactivated.

6.3.12A Storage

The space tug will remain dormant until required for a subsequent mission.

Part B – Space Tug No. 2 (Expendable)

6.3.5B Escape Burn

After space tug No. 1 separation, space tug No. 2s main engine will be ignited and will burn until the proper escape velocity has been attained.

6.3.6B Undocking

After engine shutdown, the vehicle attitude will be held constant until space tug/payload separation.

6.3.7B Coast

After separating from the payload, the space tug will remain in the coast mode. Maneuvers may be required to gain separation distance between the tug and payload to assure no contact.

6.3.8B Safing

After the desired space tug maneuvers have been accomplished, the tug mission will be terminated by safing the vehicle and allowing it to go into an uncontrolled expendable mode.

6.4 INTERFACE REQUIREMENTS

The requirement exists for communications with the tug when it is active during a planetary probe to allow target update and status checks. The communications necessary to fulfill this requirement are tracking, command, and telemetry.

Implementation should be via a USB-type system interfacing with Goldstone, Madrid. and/or Honeysuckle to Mission Control.

Space tug No. 1 requires an interface with the space station. The communications must include command, voice and telemetry to allow checkout, activation and repair. Therefore, this interface should be a USB-type which will satisfy all requirements.

6.5 MISSION TIMELINE

PHASE	BOTH SPACE TUG STAGES	DURATION	MAXIMUM ACCUMULATIVE TIME: SPACE TUG STAGE NUMBER 1	MAXIMUM ACCUMULATIVE TIME: SPACE TUG STAGE NUMBER 1
Activation		1-2 hours	2 hours	2 hours
Orbital Coast		2-3 hours	5 hours	5 hours
Transfer Burn		6-12 minutes	5.2 hours	5.2 hours
Undocking		30 seconds	5.2 hours	5.2 hours
	SPACE TUG STAGE NUMBER 1			
Orbital Coast		0.5-2 hours	7.2 hours	
Transfer Burn		4-8 minutes	7.3 hours	
Orbital Coast		2-4 hours	11.3 hours	
Transfer Burn		0.5-3 minutes	11.4 hours	
Rendezvous		2-8 hours	19.4 hours	
Dock		0.5-2 hours	21.4 hours	
Deactivate		1-2 hours	23.4 hours	
Storage		30-180 days	180 days max	180 days max
	SPACE TUG STAGE NUMBER 2			
Escape Burn		6-12 minutes		5.3 hours
Undocking		30 seconds		5.3 hours
Coast		0.5-1 hour		6.3 hours
Safing		0.5 hour		6.8 hours max

7.0 REUSABLE NUCLEAR SHUTTLE (RNS) EARTH/MOON MISSION

7.1 GUIDELINES

Vehicle: Reusable nuclear shuttle (RNS). RNS control is provided by space tug astrionics.

Payload: Space station modules, space tug, propellant, crew, miscellaneous payload.

Mission: Transfer of payload to and from the moon.

Frequency of Operation: 2 – 8 round trips per year.

Operation: Manned or unmanned. Automatic, remote operation or manual control.

7.2 MISSION DESCRIPTION

The nuclear shuttle mission, using the reusable nuclear shuttle vehicle, starts from a configuration that has the nuclear shuttle docked to a propellant and maintenance depot positioned in a circular earth orbit. The depot maintains a constant distance between it and an earth space station, so that radiation exposure to the space station is tolerable. A lunar space station in a circular, polar lunar orbit is the destination for a payload.

In preparation for the mission, the nuclear shuttle undocks from the depot and moves into position to receive the payload from a space tug plying between the space station and the depot. The space tug docks with the nuclear shuttle and transfers the payload, and the tug moves the shuttle into a lower orbit. The space tug then returns to the space station, and the nuclear shuttle proceeds to the moon.

The nuclear shuttle (see Figure 7-1) transports the payload from an earth orbit to a lunar orbit, exercising the proper maneuvers to place it at a constant distance from a lunar space station orbiting the moon (see Figure 7-2). A space tug from the lunar space station docks with the nuclear shuttle, removes the payload and transfers it to the lunar space station.

If the nuclear shuttle is to remain in the vicinity of the moon for an extended period, it maneuvers into a lunar parking orbit position which is removed from the space station. When it is to return to an earth orbit, it accepts a payload from the space tug and maneuvers into a transearth trajectory, exercising the appropriate maneuvers to place it at a constant distance from the orbiting propellant and maintenance depot.

The space tug from the earth space station docks with the nuclear shuttle, removes the payload, and returns to the space station. The nuclear shuttle docks with the depot and is prepared for its next mission.

7.3 DETAILED MISSION PHASE DESCRIPTIONS

Initial Conditions

The propellant and maintenance depot (PMD) will be station keeping with earth space station at a fixed distance.

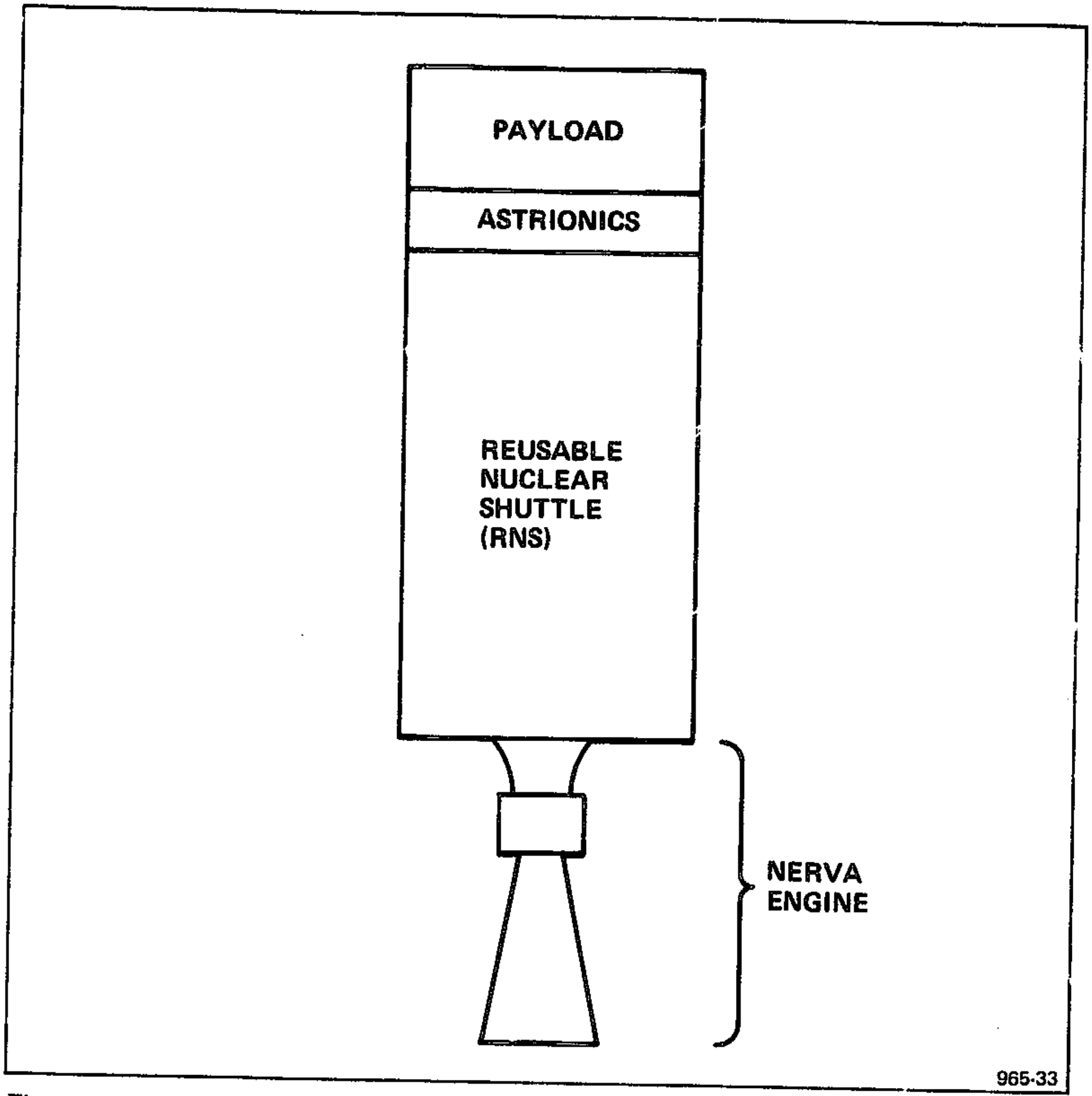

Figure 7-1. Basic Configuration of the RNS Vehicle

Nuclear shuttle will be docked with propellant and maintenance depot.

The aft interstage of the nuclear shuttle will be jettisoned.

7.3.1 Reactivation

The reactivation sequence of the nuclear shuttle will be initiated by either the propellant and maintenance depot personnel or by commands from the space station.

The automatic checkout sequence will also be initiated by either propellant and maintenance depot personnel or by commands from the space station.

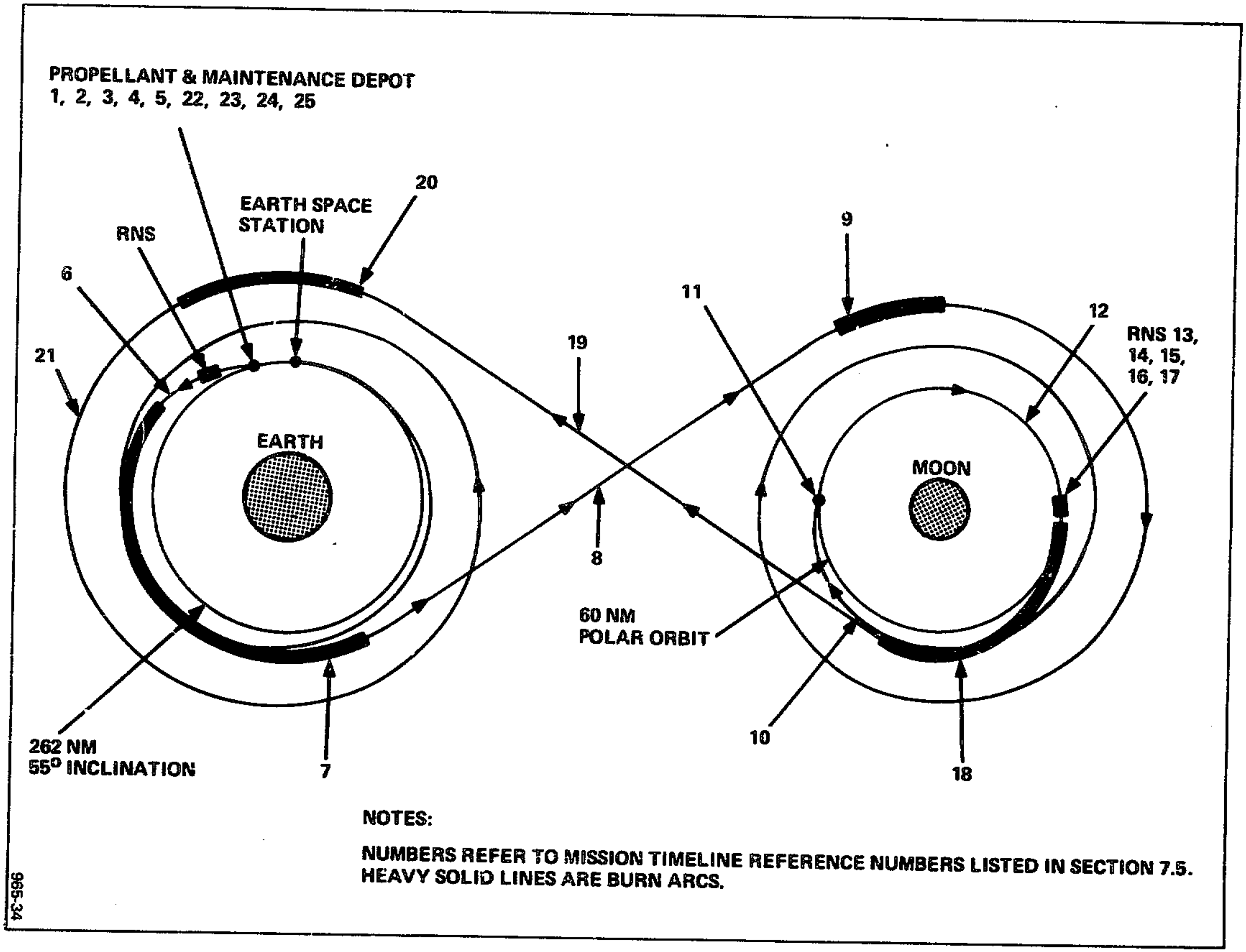

Figure 7-2. Mission Profile for Earth-Moon Reusable Nuclear Shuttle Mission

7.3.2 Maintenance

Any necessary maintenance will be performed during this phase.

7.3.3 Refueling

The nuclear shuttle will be fueled commensurate with the mission to be performed.

7.3.4 Undocking (RNS from PMD)

The earth space station will command the unmanned nuclear shuttle to undock from the depot.

If a payload is to be taken to the moon, the space tug will dock with payload and prepare to move it to the nuclear shuttle.

The nuclear shuttle will maintain a controlled attitude.

7.3.5 Docking (Space Tug with RNS)

Space tug will move payload to the nuclear shuttle and dock with it.

7.3.6 Pre-Operational

The interface between a manned payload and nuclear shuttle will be checked out, if applicable. Any necessary vehicle checkouts will also be accomplished.

The earth space station will transfer necessary data to the nuclear shuttle's astrionic system to perform lunar mission. This data includes such items as navigation updates, time to initiate operations for lunar targeting, etc.

The space tug, while docked with the nuclear shuttle will move it into a lower orbit and circularize it there. The attitude control of the nuclear shuttle will be inactive during this transfer.

The space tug will undock from the nuclear shuttle and return to earth space station.

The attitude control of the nuclear shuttle stabilizes it. The NERVA engine is not operated until a safe distance exists between the nuclear shuttle and earth space station such that the radiation level received at the earth space station is within tolerances.

7.3.7 Escape Burn (TLI)

The NERVA engine will burn for approximately 25 minutes at 75,000 pounds thrust and 825 second nominal specific impulse.

The nuclear shuttle performs the proper maneuvers to inject into a translunar trajectory which will satisfy lunar targeting. Maneuver will be performed such that the radiation level received at the earth space stations is within tolerances.

7.3.8 Transfer Coast (Translunar)

After the NERVA engine burns for approximately 25 minutes, the nuclear shuttle will experience periodic cooldown thrusts for approximately 35-50 hours. The periodic thrust will be used as efficiently as possible to provide final trajectory refinements.

The translunar coast phase will require 108 hours (Reference A-1).

7.3.9 Transfer Burn (LOI)

The NERVA engine will burn approximately 6 minutes to brake the nuclear shuttle into lunar orbit. The burn will be accomplished to properly phase the nuclear shuttle with the lunar space station.

7.3.10 Rendezvous (RNS with LOSS)

The periodic engine cooldown thrust, which will last approximately 11 hours, will be used as efficiently as possible to rendezvous the nuclear shuttle with the lunar space station. A minimum distance must be maintained between the nuclear shuttle and the lunar space station.

The nuclear shuttle will control its attitude such that the radiation received by the lunar space station and lunar surface base are within tolerances.

7.3.11 Docking/Undocking (Space Tug with RNS)

The nuclear shuttle will maintain a minimum distance from the space station. The space tug will undock from the lunar space station and will dock with the nuclear shuttle.

The space tug removes the payload from the nuclear shuttle and transfers it to the lunar space station.

7.3.12 Transfer Burn

Lunar space station will command the nuclear shuttle to change its altitude using the reaction control system.

The nuclear shuttle will coast at the new altitude until sufficient distance from the lunar space station has been attained.

The nuclear shuttle will perform, either automatically or on command from Earth, a maneuver using the reaction control system to return it to the same altitude as the lunar space station, if this maneuver is required for phasing or storage.

If extended lunar stay is planned reference Paragraph 7.3.13. If immediate earth return is planned reference Paragraph 7.3.17.

7.3.13 Deactivate

The nuclear shuttle will automatically perform a deactivation sequence to reach a semi-active state of minimum attitude control and monitoring.

7.3.14 Storage

The nuclear shuttle shall remain in this semi-active state for not more than 30 days.

7.3.15 Reactivation

The nuclear shuttle will automatically perform a reactivation sequence upon command from the earth or the space tug.

The nuclear shuttle will automatically perform a checkout sequence when the reactivation sequence is completed and communicate its condition to earth or the space tug.

7.3.16 Docking/Undocking (Space Tug with RNS)

Space tug with the payload will undock from the lunar space station and maneuver to rendezvous and dock with the nuclear shuttle.

The space tug will transfer the payload to the nuclear shuttle.

The space tug will undock from the nuclear shuttle and maneuver to return to the lunar space station.

7.3.17 Preoperational

The interface between a manned payload and the nuclear shuttle will be checked out, if applicable. Any necessary vehicle checkouts will also be accomplished.

Earth space station will transfer necessary data to the nuclear shuttle's astrionic system to perform the earth return trip. This includes such items as navigation update, time to initiate operations for earth targeting, etc.

7.3.18 Escape Burn

The NERVA engine will burn for approximately 5 to 7 minutes at 75,000 pounds thrust and 825 second nominal specific impulse.

The nuclear shuttle performs the proper maneuvers to inject into a transearth trajectory. Maneuvers will be performed so that the radiation level received at the lunar space station and lunar surface base are within tolerances.

7.3.19 Transfer Coast (Transearth)

After the NERVA engine is shutdown, the nuclear shuttle will experience a periodic cooldown thrust for approximately 11 hours. The periodic thrust will be used as efficiently as possible to provide final trajectory refinements.

The transearth coast phase will require 72 hours (Reference A-1).

7.3.20 Transfer Burn (Earth Braking)

The NERVA engine will burn approximately 8 to 12 minutes to brake the nuclear shuttle into earth orbit.

The burn will be accomplished to properly phase the nuclear shuttle with the earth space station.

7.3.21 Rendezvous (RNS with EOSS)

The periodic engine cooldown thrust, which will last approximately 13 to 20 hours, will be used as efficiently as possible to rendezvous with the earth space station.

A minimum distance must be maintained between the nuclear shuttle and the earth space station.

The nuclear shuttle will control its attitude such that the radiation received at the earth space station is within tolerances.

7.3.22 Docking/Undocking (Space Tug with RNS)

The nuclear shuttle will maintain a station-keeping position with the propellant and maintenance depot.

The space tug will undock from earth space station and dock with the nuclear shuttle.

The space tug with the payload will undock from the nuclear shuttle and dock with the earth space station.

7.3.23 Docking (RNS with Depot)

The nuclear shuttle will automatically dock with the propellant and maintenance depot using the reaction control system.

7.3.24 Maintenance

The propellant and maintenance depot personnel will perform the required maintenance actions.

7.3.25 Deactivation

The earth space station will command the nuclear shuttle to automatically perform a deactivation sequence.

7.3.26 Storage

The nuclear shuttle will remain in the deactivated state until needed for its next mission.

7.4 INTERFACE REQUIREMENTS

The RNS mission will require an interface with the space station, tug, and the LOSS to allow targeting updates during rendezvous and the transfer of data for status purposes when in line-of-sight of these space vehicles. During translunar coast an interface with the ground for the same reasons is required.

The system installed on the RNS should be a USB type system including command, tracking and TM. The ground systems will be Madrid, Goldstone, and Honeysuckle with the normal data relay to Mission Control.

7.5 MISSION TIMELINE

	PHASE	DURATION (DAYS)	CUMULATIVE TIME (DAYS)
	1. Activation/Reactivation		
	2. Maintenance		
	3. Fueling/Refueling		
	4. Undocking (RNS from Depot)	7	7
(1)	5. Docking (Space Tug with RNS)		
	6. Pre-operational		
	7. Escape Burn (TLI)		
	8. Transfer Coast (Translunar)	4	11
	9. Transfer Burn (Lunar Braking)		
	10. Rendezvous (RNS with Lunar Space Station		
	11. Docking/Undocking (Space Tug with RNS		
	12. Transfer Burn (Lunar Orbit Change)		
(2)	13. Safing (Deactivation)	Max. 30 (Assumes 28 day lunar orbit storage); Min. 2	Max. 41; Min. 13
(2)	14. Storage		
(2)	15. Reactivation		
(3) (2)	16. Docking/Undocking (Space Tug with RNS)		
	17. Pre-operational		
	18. Escape Burn (TEI)		
	19. Transfer Coast (Transearth)		Max. 45; Min. 17
	20. Transfer Burn (Earth Braking)	4	
	21. Rendezvous (RNS with Earth Space Station)		
(3)	22. Docking /Undocking (Space Tug with RNS)		
	23. Docking (RNS with Depot)		
	24. Maintenance		
	25. Safing (Deactivation)	Variable	
	26. Storage		

(1) This phase applicable only if payload is to be carried to moon.
(2) This phase applicable only if extended lunar stay time is planned.
(3) This phase applicable only if payload is to be returned to earth.

8.0 LUNAR LANDING MISSION

8.1 GUIDELINES

Vehicle: Reusable space tug (landing leg kit, propulsion module, astrionic module, cargo module, crew module, and manipulator arm kit).

Mission: Manned or unmanned lunar landing and return to lunar orbit space station.

Launch Vehicle: Tug brought to lunar orbit space station by RNS or payload on Saturn V.

Storage: Up to 180 days quiescent storage in lunar orbit. Up to 42 days on lunar surface. Operational within two hours.

Operation: Primarily manned, but capable of automatic or remote operation.

Frequency of Operation: Alternately, 28 days on lunar surface with 28 days docked to space station.

8.2 MISSION DESCRIPTION

The space tug mission will begin with the tug configured for a lunar landing mission, in a quiescent mode, and docked to the lunar space station (60 NM polar lunar orbit altitude).

The space tug will be activated, checked out to assure operational capability of all subsystems and components, and onboard systems initialized with mission parameters. The space tug will undock from the lunar space station when the station is properly phased with the prospective landing site. The space tug will transfer to a 60 NM by 9 NM lunar orbit. At perilune (9 NM), the space tug will initiate a powered descent burn to the lunar surface.

After landing, the space tug will spend 14 to 28 days on the lunar surface. The space tug systems required for lunar habitation and exploration will be activated, and any astrionic subsystems not required during the lunar exploration will be deactivated.

After lunar surface exploration, the space tug astrionic systems will be activated, and the space tug will inject into an initial 9 x 45 NM lunar orbit for subsequent phasing maneuvers with the lunar space station. The space tug will rendezvous and dock with the space station, and the space tug will be deactivated and stored. The mission profile and vehicle configuration are shown in Figure 8-1.

Prior to the next use of the space tug for lunar landing, the space tug subsystems will be thoroughly checked out and defective parts replaced.

8.3 DETAILED MISSION PHASE DESCRIPTIONS

Initially, it is assumed that the space tug used for the lunar landing will be docked to the lunar orbit space station (LOSS). The LOSS will be in a 60 NM circular polar orbit. The space tug will be powered down to the maximum extent.

The following are the detailed mission phases for this mission.

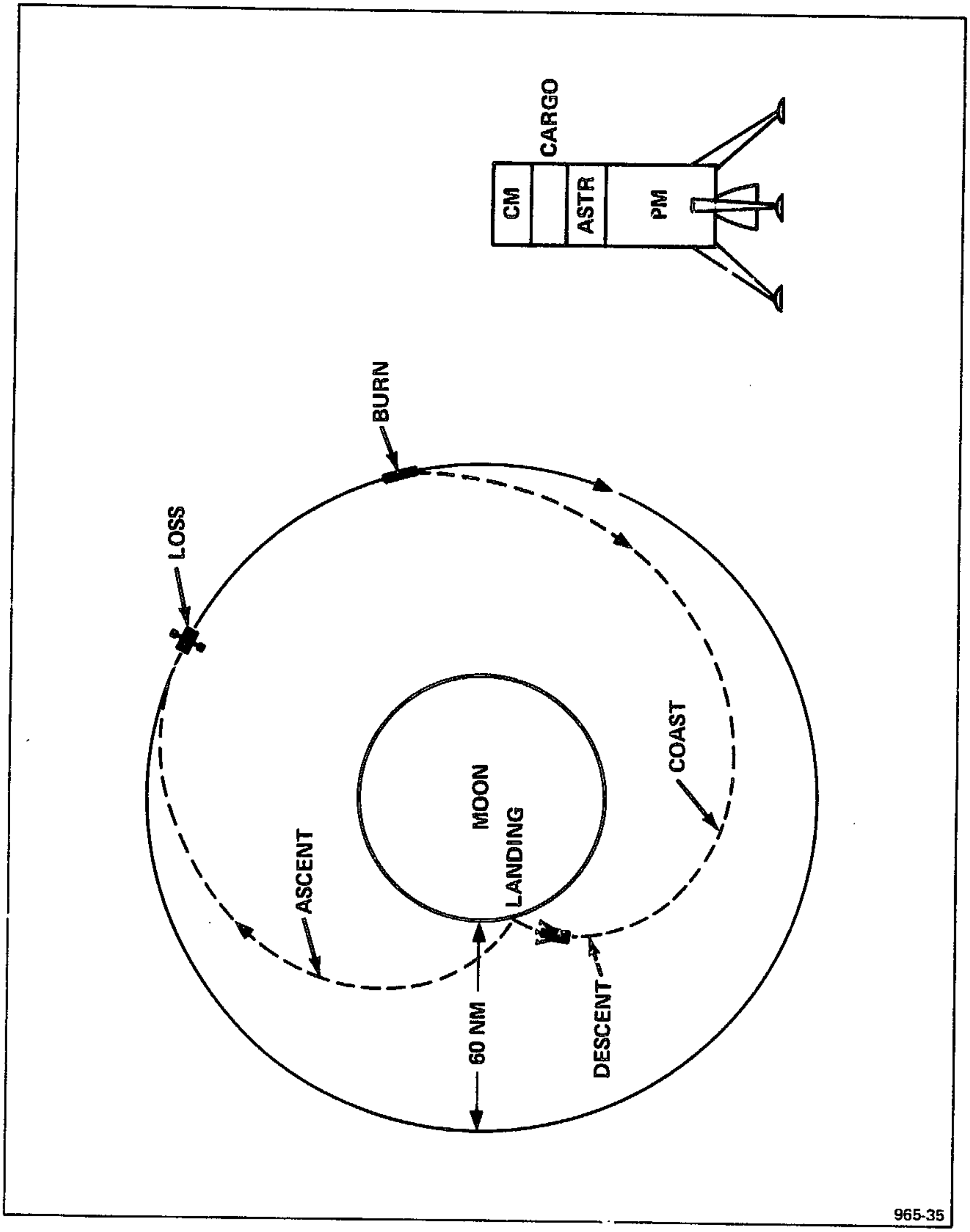

Figure 8-1. Lunar Landing Mission Profile and Vehicle Configuration

8.3.1 Activation Phase

The space tug will be activated, and all onboard astrionic subsystems will be checked out. When all onboard systems are deemed operational, the onboard systems will be initialized with the mission parameters.

8.3.2 Undocking Phase

The space tug will undock from the space station and maneuver a safe distance away from the space station to begin its transfer burn. During this phase, the space tug will maneuver while a member of the space station crew visually checks the vehicle.

8.3.3 Transfer Burn

When the space tug is a safe distance from the space station and approximately 180° from the landing site, the initial retrograde burn of the space tug will be used to place the space tug in a 9 NM by 60 NM elliptical orbit. The burn time will be from 20 to 40 seconds in duration.

8.3.4 Orbital Coast Phase

The space tug will coast for either one-half revolution or 1.5 revolutions until the initiation of powered descent at perilune. During this phase, the onboard systems will be monitored and configured for the next burn.

8.3.5 Powered Descent and Landing Phase

At perilune, the powered descent to the lunar surface will begin. This phase will begin at 9 NM altitude and continue until the space tug has landed on the lunar surface. The tug will be controlled automatically or by remote operation. This phase is assumed to be similar with the present lunar landing mission. The burn time will be from 10 to 14 minutes duration.

8.3.6 Deactivation Phase

After landing, the onboard systems will be checked to detect any possible damage incurred during the landing. The subsystems required to maintain survivability of the crew and/or the astrionic systems will be partially activated. All other subsystems will be deactivated.

8.3.7 Storage Phase

The space tug will remain in the deactivated (or quiescent) state for the duration of the lunar surface stay. The onboard detection system will detect and report any critical malfunctions or off-nominal conditions during the storage phase. This phase will last either 14 or 28 days, with an additional 14 days contingency capability required.

8.3.8 Reactivation Phase

After the lunar surface exploration, the space tug will be reactivated, the onboard systems will be checked out and the space tug configured for the ascent burn.

8.3.9 Powered Ascent Phase

A 6 to 10 minute burn of the space tug will place the tug in a 9 NM by 45 NM elliptical orbit suitable for rendezvous with the lunar orbit space station. Part of the space tug may be expendable and left on the lunar surface. The powered ascent is assumed to be similar to the present powered ascent by the lunar module on the Apollo missions.

8.3.10 Orbital Coast Phase

The space tug will coast until apolune where a burn will be performed to circularize the orbit at 45 NM. During this phase, the tug will update or receive data to update its trajectory for rendezvous with the space station.

8.3.11 Rendezvous Phase

After the coast, a burn will circularize the orbit at 45 NM. The space tug will coast in this orbit until combinations of short burns and coast periods maneuver the tug into proper position for docking with the space station. The present Apollo rendezvous scheme, after circularizing at 45 NM, includes a plane change maneuver (if required), a burn one-half orbit after circularizing to provide a constant delta height between the rendezvousing vehicles, a two-burn Hohmann transfer to approximately the final orbit, and the final close-range rendezvous and docking. These maneuvers will place the tug in a 60 NM circular orbit with the space station.

8.3.12 Docking Phase

When the tug maneuvers to a close proximity of the space station, the space tug will dock with the space station (either manually or by remote operation, if unmanned).

8.3.13 Deactivation Phase

With the space tug docked to the lunar orbit space station, the space tug will be given a final status check and the onboard systems powered down. The space tug may receive external power and/or environmental conditioning from the space station.

8.3.14 Storage Phase

The tug will be placed in a quiescent status and remain docked to the space station until fueling and maintenance is complete or the next active mission is started.

8.3.15 Maintenance Phase

During some or all phases of the mission, the tug will be checked out. If defective modules are found during this checkout, they will be replaced at the next appropriate and convenient time. Periodic maintenance will be performed. This would be performed during slack periods, and any defective modules or units would be replaced and the subsystems returned to optimum operating capability. The frequency of maintenance will depend on the number of missions flown, time in space, past failure rates, etc.

8.3.16 Fueling Phase

The space tug will be activated for fueling when its fuel supply is exhausted. This may be done randomly, such as when the space shuttle brings fuel, or scheduled just prior to a mission. The tug will actively monitor the fueling process.

8.4 INTERFACE REQUIREMENTS

A USB system should be installed in the astrionic module to allow communications with the ground system at Goldstone, Honeysuckle, and/or Madrid for transmission of voice, data and command between the NASA Mission Control and the tug while on the lunar surface.

A similar communications link is required between the LOSS and the tug for command, control and evaluation.

The ground link is required because the LOSS will not always be within line-of-sight of the tug when the tug is on the lunar surface.

8.5 MISSION TIMELINE

PHASE	DURATION	MAXIMUM ACCUMULATIVE TIME
Activation	2 hours	2.0 hours
Undocking	0.5 hours	2.5 hours
Transfer Burn	20 – 40 seconds	2.5 hours
Orbital Coast	1 – 3 hours	5.5 hours
Powered Descent and Landing	0.2 hours	5.7 hours
Deactivation	4 hours	9.7 hours max.
Storage	14 – 28 days	42 days max
Reactivation	2 hours	2.0 hours
Powered Ascent	.1 – .2 hours	2.2 hours
Orbital Coast	1 – 4 hours	6.2 hours
Rendezvous	2 – 8 hours	14.2 hours
Docking	0.5 hours	14.7 hours
Deactivation	4.0 hours	18.7 hours max.
Storage	14 – 28 days	180 days

9.0 FOUR STAGE SATURN V MISSION

9.1 GUIDELINES

Vehicle: Saturn V (without IU) with the space tug as a fourth stage.

Payload: Lunar orbit space station, space tugs, CSM derivative, fuel, etc.

Mission: Transfer of payload from earth to lunar orbit.

Frequency of Operation: Will be used in lunar orbit after completion of this mission. Frequency or length of deactivation is unknown.

Operation: Automatically controlled; however, a crew may have control capability.

9.2 MISSION DESCRIPTION

The Saturn V/space tug configuration will be launched from Complex 39. Complete burns of the S-IC and S-II and a partial burn of the S-IVB will insert the configuration into a 100 NM circular parking orbit. After two to three orbits, the S-IVB will be reignited for the transfer burn into a lunar trajectory. The S-IVB will burn to depletion; it will be jettisoned; and the space tug fourth stage will complete the lunar transfer burn.

The space tug/payload will coast approximately three days in the translunar coast. Any midcourse corrections will be performed by the space tug.

When the space tug/payload reaches the 60 NM perilune behind the moon, the space tug, in a retrograde attitude, will fire its engine(s) to insert the payload into a 60 NM by 170 NM lunar orbit. A second lunar orbit insertion burn will insert the payload into a 60 NM circular polar orbit in the vicinity of the lunar space station. After another space tug removes the payload from the space tug (fourth stage), the fourth stage will be jettisoned unless it is required for use with the lunar space station.

The vehicle configuration and mission profile are shown in Figure 9-1.

9.3 DETAILED MISSION PHASE DESCRIPTIONS

9.3.1 Preflight Operations

Launch pad checkout activities will be conducted for S-IC, S-II, S-IVB, and space tug stages prior to launch to assure that all systems are capable of operating as planned.

9.3.2 Boost-to-Orbit (To LEO)

The vehicle will be targeted for a 100 NM circular earth parking orbit. It will attain this orbit by complete burns of the S-IC and S-II stages and a partial burn of the S-IVB stage.

9.3.3 Orbital Coast (LEO)

After S-IVB stage cutoff the vehicle will enter the coast mode for one to three earth orbits. During this coast period, additional vehicle checks will be made to assure that all systems are capable of proper operation prior to TLI.

9.3.4 Translunar Injection (TLI) Burn

The escape burn to a lunar trajectory shall be made by burning the S-IVB stage to depletion, separating the S-IVB stage, and igniting the space tug fourth stage for a partial burn. When proper cutoff conditions have been attained, the space tug engine will cut off. The S-IVB stage will provide a delta V after separation to avoid contact with other space vehicles and will be disposed of in an escape orbit, if possible.

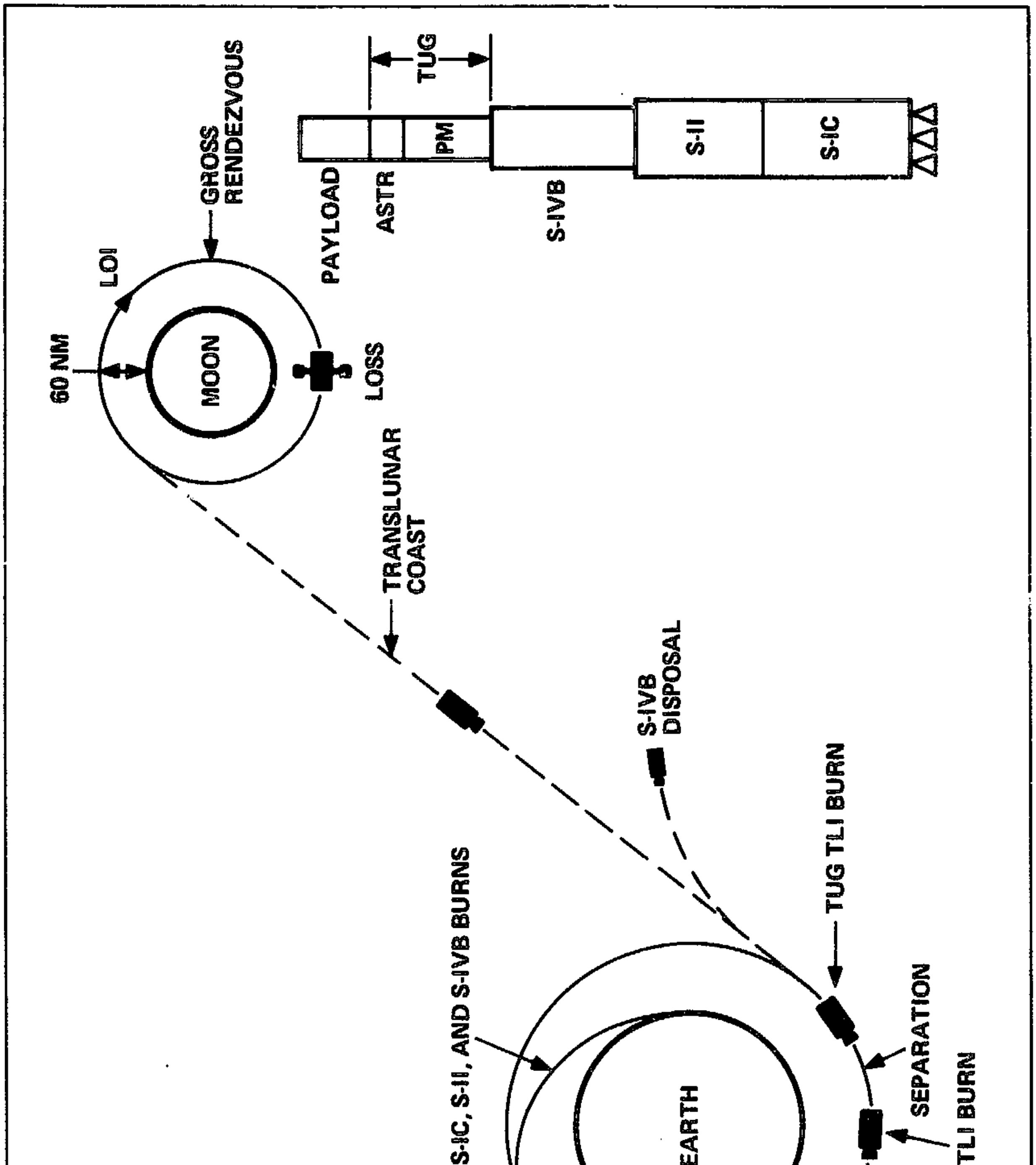

Figure 9-1. Space Tug Four Stage Saturn V Mission Profile and Vehicle Configuration

9.3.5 Translunar Coast

The fourth stage will coast for approximately three days to reach the moon. Any required midcourse corrections will be performed by the space tug. The tug will be rolled to equalize the solar heating effect. The ground will aid in target and navigation updates.

9.3.6 Transfer Burn No. 1 (LOI)

Upon lunar arrival, the space tug, in a retrograde attitude, will fire its engine for insertion into a 60 NM by 170 NM elliptical polar orbit.

9.3.7 Orbital Coast (Lunar)

The space tug will remain in the 60 NM by 170 NM orbit for phasing with the lunar orbit space station (LOSS). The coast phase will last one to three revolutions.

9.3.8 Transfer Burn No. 2 (LOI)

A second retrograde burn of the space tug will circularize the payload in an approximately 60 NM circular polar orbit in close proximity with the LOSS.

9.3.9 Rendezvous (Tug with LOSS)

After the circularization burn, the space tug will maneuver to accomplish rendezvous with the LOSS. The space tug can now dock its payload to the LOSS, or its payload may be removed by another lunar space tug.

The space tug fourth stage mission is now complete; however, all or portions of the tug may be subsequently used for lunar operational activities.

9.4 INTERFACE REQUIREMENTS

This mission, from a support standpoint, is the same as the lunar mission and should use the MSFN while in low earth orbit and use Madrid, Goldstone, and Honeysuckle in the translunar coast and lunar orbit phases.

To implement this interface a USB type system should be installed on the tug with voice, command, TM, and tracking capability.

9.5 MISSION TIMELINE

PHASE	DURATION	MAXIMUM ACCUMULATIVE TIME
Preflight	Up to 20 hours	20 hours max.
Boost-to-Orbit	10–14 minutes	0.2 hours
Orbital Coast	.7–4.5 hours	4.7 hours
Translunar Injection Burn	6–10 minutes	4.9 hours max.
Translunar Coast	2.5–3.5 days	3.5 days max.
Transfer Burn No. 1 (LOI_1)	5–10 minutes	0.2 hour
Orbital Coast	1–6 hours	6.2 hours
Transfer Burn No. 2 (LOI_2)	2–6 minutes	6.3 hours
Rendezvous	1–4 hours	10.3 hours max.

REFERENCES

A-1. Guidelines and Constraints Document, Nuclear Shuttle Systems Definition Study, Phase A, Revision No. 1, May 28, 1970, MSFC Document No. PD-SA-P-70-63.

APPENDIX B

PRELIMINARY ASTRIONIC SYSTEM FUNCTIONAL REQUIREMENTS FOR SPACE TUG MISSIONS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION AND SCOPE

The astrionic system, by definition, includes that integrated group of components and/or subsystems which provide the following vehicle functions during a mission:

- Navigation, guidance, and control of the vehicle
- Measurement of vehicle parameters
- Onboard data management
- Data transmission between vehicle, ground stations, and other space systems
- Facilitate vehicle tracking by ground stations or other space systems
- Checkout and monitoring of vehicle functions
- Detection of emergency situations
- Generation of electrical power for system operation
- Power and signal distribution
- Thermal conditioning of components

Current practice shows the astrionic system to be at level 5 in the NASA work breakdown structure used for determining costs of new programs.

This appendix defines the preliminary astrionic system functional requirements for seven space tug missions. These seven missions were selected as the "design" missions which determine the tug astrionic system. Descriptions of these missions are provided in the preceding appendix and are summarized in Figure 1-1.

The space tug is of modular design, and vehicle groundrules dictate that the primary astrionic equipment be assembled into a structural module or segment called the astrionic module. This module is at level 4 in the NASA work breakdown structure. Functions and/or equipment external to the astrionic module is to be defined to the extent needed to size and define interfaces and functional requirements of the astrionic system.

2.0 SUMMARY OF RESULTS

Analysis of projected availability dates for the major NASA space systems and a scheduling of the seven design missions indicates that the capability to satisfy the whole range of space tug astrionic system functional requirements must be provided within a two year time frame.

Derivation of preliminary space tug astrionic system functional requirements included definition of the design criteria and constraints which were assembled from the mission operations and vehicle descriptions, study guidelines, system interfaces, and general effectiveness and safety requirements as well as definition of typical performance limits for the astrionic system functions required in each mission phase.

LEGEND:

R&D DENOTES RENDEZVOUS AND DOCK
P/L DENOTES PAYLOAD
AST. DENOTES ASTRIONICS
PROP. DENOTES PROPULSION

MISSION	VEHICLE MANNED (M), UNMANNED (U), OR BOTH (B).	PAYLOAD MANNED (M), CARGO (C), OR BOTH (B).	VEHICLE AND/OR TUG MODULES	NOTES
1. SYNCHRONOUS ORBIT (REUSABLE) – TRANSFER P/L FROM LEO TO EQUATORIAL SYNCHRONOUS ORBIT, R&D, RETURN P/L TO LEO, AND R&D.	U	U	TWO TUGS IN TANDEM 1ST TUG – AST. & PROP. MODULES 2ND TUG – AST., CARGO AND PROP. MODULES	• TUGS TRANSPORTED TO LEO BY EOS • ASSEMBLY OPERATIONS PERFORMED IN LEO • POSSIBILITY OF MANNED MISSIONS EXIST
2. SYNCHRONOUS ORBIT (EXPENDABLE) – TRANSFER P/L FROM LEO TO EQUATORIAL SYNCHRONOUS ORBIT, R&D, AND DISPOSE OF TUG.	U	U	TUG AST., CARGO, AND PROP. MODULES	• TUG TRANSPORTED TO LEO BY EOS
3. ORBITAL OPERATIONS (EARTH OR LUNAR ORBIT) – TRANSFER FROM SPACE STATION TO LOWER ORBIT, R&D, LOAD P/L, TRANSFER UP TO SPACE STATION, AND R&D.	B	B	TUG CREW, AST., CARGO, AND PROP. MODULES	• TRANSPORTED TO LEO BY EOS OR SAT. V • OPERATIONAL BASE IN ORBIT AVAILABLE • TRANSPORTED TO LUNAR POLAR ORBIT USING 4 STAGE SAT. V • CAPABILITY OF OPERATION BY REMOTE CONTROL REQUIRED
4. PLANETARY MISSION (TWO TUGS, ONE EXPENDED) – FIRST TUG PROVIDES INITIAL THRUST FOR INJECTION AND RETURNS TO BASE. SECOND TUG PROVIDES REMAINING THRUST FOR EARTH ESCAPE.	U	U	TWO TUGS IN TANDEM. BOTH TUGS HAVE AST. AND PROP. MODULES.	• TUGS TRANSPORTED TO LEO BY EOS • ASSEMBLY OPERATIONS PERFORMED IN LEO
5. RNS EARTH-MOON MISSION – TRANSFER P/L BETWEEN LEO AND LUNAR POLAR ORBIT.	B	B	REUSABLE NUCLEAR SHUTTLE	• PROPELLANT AND MAINTENANCE DEPOT IN LEO • DOCKING AND UNDOCKING TO BE CONTROLLED FROM SPACE STATION (UNMANNED RNS)
6. LUNAR LANDING – TRANSFER P/L BETWEEN LUNAR POLAR ORBITING SPACE STATION AND LUNAR SURFACE.	B	B	TUG CREW, AST., CARGO, AND PROP. MODULE PLUS KITS.	• TRANSPORTED TO LUNAR SPACE STATION BY RNS • QUIESCENT STORAGE OF 180 DAYS IN ORBIT AND UP TO 42 DAYS ON SURFACE
7. FOUR STAGE SATURN V TRANSFER P/L FROM EARTH BASE TO LUNAR POLAR ORBIT.	B	B	SATURN V WITH TUG AS FOURTH STAGE.	• TUG ASTRIONICS TO PROVIDE VEHICLE NG & C ASTRIONICS FUNCTIONS • MANNED PAYLOAD TO HAVE CONTROL TAKE-OVER CAPABILITY

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Figure 1-1. Design Missions Summary Chart

Astrionic system functions are interfaced with overall space tug operations by a cursory analysis of functional flow diagrams for the tug. More work is needed to fully define these interfaces.

3.0 REQUIREMENTS ANALYSIS

3.1 DESIGN CRITERIA AND CONSTRAINTS

The astrionic system(s) design criteria and constraints were assembled and derived from the missions operations and vehicle descriptions, study guidelines, system interfaces, and general effectiveness requirements. These design criteria and constraints are divided into the following categories:

(1) Mission Mix and Schedule

(2) Operational Bases and Facilities

(3) Modularity

(4) Autonomy

(5) Interfaces

(6) Environments

(7) Effectiveness

(8) Monitor and Control

(9) Storage

(10) Operational Life

(11) Transport

3.1.1 Mission Mix and Schedule

The seven design missions, as shown in Figure 1-1, encompass fifteen different mission-vehicle-payload combinations. These combinations include three different vehicles with three configurations of one of the vehicles (tug), manned and unmanned vehicles, single mission and reusable vehicles, and payloads that are cargo and manned. From Figure 1-1, it should be noted that all seven representative missions must be capable of being performed in an unmanned mode, i.e., no support from an onboard crew or crew module. Also all seven missions, except the deep space and synchronous orbit with expendable vehicle missions, may be manned or have manned payloads.

Figure 3-1 illustrates the projected availability dates of the major NASA space systems. Based on these availability dates, a scheduling of the seven representative missions shows that the whole range of space tug astrionic system functions must be made available within a 2-year time frame. This assumes an Apollo Program follow-on. If there is no Apollo Program follow-on, there will be about a 3-year gap between the earth orbit and planetary missions and the lunar missions.

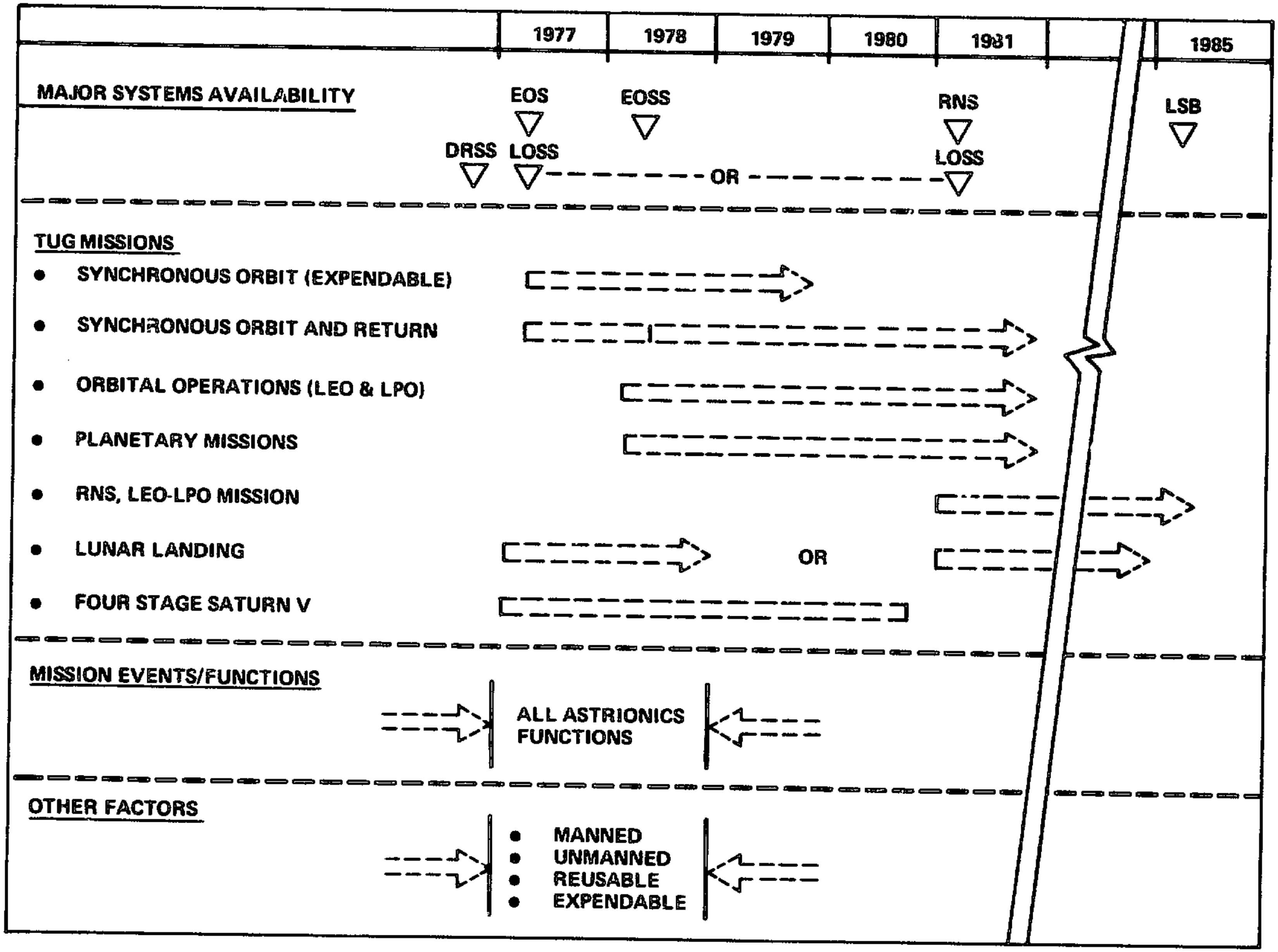

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Figure 3-1. Functional Requirements Schedule

3.1.2 Operational Base and Facilities

Each of the representative missions has a point-of-origin or operational base where the "prepare for mission" functions are performed. These operational bases include:

(a) Earth base (KSC or other) – for the early synchronous orbit (expendable vehicles) missions and for the four stage Saturn V lunar missions.

(b) Earth orbit space station – for the low earth orbital operations, planetary missions, and synchronous orbit (reusable vehicles) missions and also as maintenance depot for RNS mission support.

(c) Lunar orbit space station – for tug operations in lunar orbit and lunar landing missions.

The "prepare for mission" functions include checkout, replacement part maintenance, replace or load expendables, loading a new program and/or navigational data or parameters into the computer, and other operations, such as configure tug(s) for assigned mission (stack two tugs for tandem operation and/or add special kits), and load fuel and payload.

Use of expendables should be minimized to the extent practical. Other supplies, such as replacement parts, should be limited as to types and quantities required.

3.1.3 Modularity

To meet the functional requirements of the many mission-vehicle-payload combinations the astrionic system(s) must possess both a high degree of operational flexibility and be readily changeable through the addition or deletion of special kits. The tug is to be of modular design (crew, astrionic, cargo and propulsion modules plus special kits). The tug is to be configured in space by addition or removal of modules and/or kits for assigned mission. For some missions, stacking tugs for tandem operation is required. The astrionic module/segment for the RNS and Saturn V may be configured at KSC or other earth base for these vehicles.

The astrionics for both the tug and RNS must be capable of being reconfigured, as required, in space to provide the required mission flexibility. This reconfiguration capability should be provided by the addition and/or removal of components. This reconfiguration capability is made possible with a "modular astrionics" concept and does not require all equipment to be designed for "universal" capability.

3.1.4 Autonomy

Autonomous operations has been identified as a desirable feature to reduce ground support requirements. Substantially less ground support should be required for the tug operations than is provided for the Apollo missions. Two areas where ground support should be reduced are: (a) quantity of status data should be reduced by use of onboard diagnostics, data compaction, or by limiting transmission time, and (b) ground stations should not be required to supply data that can reasonably be supplied by onboard systems.

3.1.5 Interfaces

Communication links must be provided with all mission elements. These links should be of a USB type and provide telemetry, command, voice (if manned), tracking and TV, as required for the individual missions. Some missions will require an interface or relay capability to DRSS. For DOD missions the interface should be through the satellite control facility via a DOD satellite system.

A hardwire tie-in capability should be provided for data (data bus interface) and power interchange between the tug astrionic module and other system elements, including, EOS, EOSS, LOSS, and RNS.

Within the astrionic system(s) of all space elements, a standard interface should be used to the maximum extent practical permitting interchange of components or subsystems. This interchange of components or subsystems is required to configure astrionics for specific missions and to permit updating the astrionics by using improved or advanced state-of-the-art equipment. Modifications should be limited to software changes where practical.

3.1.6 Environments

Natural – the natural environments encountered by the astrionic systems shall be as defined in:

(1) Space – NASA TM X-53798 entitled, "Space Environment Criteria Guidelines for Use in Space Vehicle Development."

(2) Terrestrial – NASA TM X-53328 entitled, "Terrestrial Environment (Climatic) Criteria Guidelines for Use in Space Vehicle Development."

Induced – The astrionics must be capable of withstanding the induced environments, either operating or non-operating, as applicable, encountered during:

(1) Transport to launch site and/or earth base and handling.

(2) Transport from earth base to LEO by EOS and jettisoning to free flight.

(3) Transport to lunar orbit as payload in either a four stage Saturn V or RNS.

(4) Operation in either the fourth stage of the Saturn V launch vehicle or the RNS.

(5) Expected tug flight operations either as single units or in tandem.

3.1.7 Effectiveness

Reliability – The astrionic system(s) shall meet the reliability requirements when maintenance, refurbishment, or replacement are used to maintain the required level of reliability throughout its lifetime. Cost effectiveness studies will determine reliability levels and maintenance cycle.

Safety – For all manned missions the astrionic system will be designed for a fail operational – fail safe condition on the equipment required for crew safety. An emergency detection system shall be provided with appropriate crew displays, and an abort mode/sequence shall be provided for each mission and mission phase where necessary. For automated vehicles with manned payloads, a control takeover capability shall be provided to the payload.

Maintainability – Onboard checkout for the complete tug/vehicle is a design goal. Checkout to the level necessary to locate failures or degraded conditions to the lowest replacement unit consistent with the modular astrionics concept is required. Support from the operational base is provided and may consist of supplying spare parts, personnel and equipment for replacing components or an entire astrionic module and assistance in checkout as required. In-flight maintenance is to be limited to switching-in redundant equipment.

3.1.8 Monitor and Control

Mission Control (Earth, Space Station, or other locations) has a responsibility to schedule and to monitor and control mission events. Onboard equipment must be provided for interchange of data, receipt and execution of commands, and to aid ground tracking stations.

Most missions require rendezvous and docking operations. For the terminal rendezvous phase and the docking phase where the tug is unmanned and the target is manned, a tug monitor and control capability shall be provided to the target vehicle crew.

3.1.9 Storage

Between missions storage, either docked to a space station/base or free flying, is required. This storage may be either powered-down or in a quiescent state, as required, to withstand the space environments, provide periodic assurance as to operational status, and to permit the tug to go from the storage mode to operational status within two hours. Storage periods of up to 180 days are required. For the lunar landing mission, a storage or quiescent mode is required for up to 42 days (28 days plus 14 day contingency) on the lunar surface.

3.1.10 Operational Life

The reusable space tug shall have a minimum lifetime goal of ten years or the capability of being reused ten times before major refurbishment. To achieve this lifetime goal, "as-required" in-space maintenance is considered mandatory. Major refurbishment may be provided by returning either the tug or astrionic module to earth base.

The operational lifetime of the astrionic system for the RNS must, as a minimum meet the RNS life requirements (presently one year or ten missions). Astrionics maintenance support required to provide the operational lifetime should be compatible with RNS maintenance plans and facilities.

3.1.11 Transport

The tug is to be transported to an operating base using an EOS, RNS, or the four stage Saturn V. For EOS transport, the astrionic module diameter will be limited to fit within the EOS cargo bay (less than 15 feet in diameter).

3.2 LUNAR LANDING MISSION REQUIREMENTS

The lunar landing mission is used to illustrate typical astrionic system requirements for the space tug. Unique requirements of other design missions will be documented in follow-on study reports.

3.2.1 Navigation, Guidance and Control Requirements

The NG&C requirements were derived during this study and are given in the following paragraphs. Also for clarity the following definitions are used:

Navigation – Navigation is the process of determining the state (velocity and position) of the vehicle with respect to a selected coordinate system and vehicle attitude with respect to the navigation coordinate system.

Guidance – Guidance is the process of determining vehicle maneuvers, engine start and cutoff timing and engine throttling conditions required to satisfy a given set of end conditions defined as either orbital constraints or attitude constraints.

Control – The control function is to maintain vehicle stability and to execute guidance maneuvers based on attitude error and attitude rate error signals developed by the guidance system. Attitude control is accomplished by gimballing the main engine or firing RCS engines on command.

3.2.1.1 Phase Independent Requirements

Navigation

All navigation requirements are treated as phase dependent requirements in paragraph 3.2.1.2.

Guidance

All guidance requirements are treated as phase dependent requirements in paragraph 3.2.1.2.

Control

- Attitude and translational control shall be provided for multiple missions.
- Capability shall exist for the following modes of attitude and translational control for both normal and emergency operational modes of the vehicle:

(1) Manual control

(2) Remote operation

(3) Automatic control

- RCS attitude control deadbands shall be variable to at least two settings, fine and coarse.

- The desired system deadbands shall be set by remote operation, crew, or automatically.

3.2.1.2 Phase Dependent Requirements

(1) Activation/Reactivation Phase

Navigation

- Initialize the navigation subsystem with a state vector (position and velocity) received from the space station. This state vector must be to the following accuracy: Position ± 2km, Velocity ± 2m/sec.

- The attitude reference supplied by the IMU will be aligned to an accuracy of one minute of arc through the use of a star tracker located either on the IMU or on the vehicle. The star tracker measures the inertial angles of the IMU or vehicle with respect to two cataloged stars.

Guidance

There are no guidance requirements during this phase.

Control

There are no control requirements during this phase.

(2) Undocking Phase

Navigation

- Compute position, velocity and attitude with respect to the reference coordinate system through the use of the IMU to an accuracy consistent with requirements of subsequent phases.

- Maintain attitude reference to an accuracy of 10 arc minutes.

Guidance

- After undocking has occurred, provide attitude control commands to:

  (1) Stabilize the tug

(2) Transfer the tug to a station keeping position

(3) Control the tug in a station keeping position

- To minimize collision possibility after undocking:

  (1) When going to a higher altitude orbit, separate from mating vehicle in direction of present velocity vector.

  (2) When going to a lower altitude orbit, separate from mating vehicle in the opposite direction of present velocity vector.

<u>Control</u>

- Translational control shall be provided via RCS.
- The maneuver is not time critical and monitoring of precise attitude control following separation is not a requirement.
- Separation velocities of 1 ft/sec or less can be used and then nulled after separation to allow the crew to check the control system operation.
- The attitude control subsystem shall execute guidance commands which are functions of relative attitude, position, and rates of change between the tug and the lunar orbit space station.
- The attitude control subsystem deadband shall be set to TBD (tentatively $\pm 0.5^{\circ}$).
- The tug shall be oriented and stabilized in preparation for the transfer burn.

(3) Orbital Coast Phase

<u>Navigation</u>

- Accept navigation update and store in computer.
- Align IMU prior to each navigation update.

During orbital coast, prior to transfer burn:

- Update the tug state vector by using a landmark tracker and orbital altimeter (radar or laser) as inputs to a Kalman filter navigation scheme or from the MSFN tracking network. The accuracy of the update will be: Position $\pm$ 2km, Velocity $\pm$ 2 m/sec.
- IMU alignment should be performed within one hour of any powered maneuver in order to maintain an accurate (10 min of arc) inertial reference during the maneuver.

Guidance

- Provide attitude control commands to:

  (1) Control the tug's attitude during the coast phase.

  (2) Orient the tug and maintain attitude hold for engine firing for the transfer burn.

Control

- A minimum impulse limit cycle shall be established in all three axes with a wide or narrow attitude deadband.

- The wide deadband shall nominally be equal to TBD (tentatively $\pm 5.0^{\circ}$) and the narrow deadband shall nominally be TBD (tentatively $\pm 0.5^{\circ}$) about each axis.

- RCS thruster commands shall be such that the number of thruster operations shall be minimized, and the frequency of thruster operation shall be less than TBD (tentatively 7 pulses per second).

(4) Transfer Burn

Navigation

When the angle between the plane of the tug orbit and the vector from the center of the moon to the landing site is within a predetermined angle and the central angle between the landing site and the tug is between 170 and 190 degrees, the tug will burn into a 9 NM by 60 NM elliptical orbit.

- Measure transfer burn parameters.

- Maintain attitude reference to an accuracy of $\pm$ 10 arc minutes.

Guidance

- Issue commands to start and shutdown the main engine and to control the direction of vehicle thrust to achieve a 9 NM by 60 NM elliptical orbit.

Control

- TVC shall provide pitch and yaw attitude control.

- Engine gimbal position and rate shall not exceed TBD (tentatively $\pm 6.0^{\circ}$ and $\pm 0.2^{\circ}$/sec, respectively).

- The TVC engine shall not respond to frequencies exceeding TBD Hz. The break frequency of the actuator should occur at a frequency of 8 to 10 times the control frequency of the system.

- The attitude control system shall respond to guidance commands within the design limits of the vehicle and engine requirements.

- Roll control shall be provided by RCS.

(5) Powered Descent and Landing

Navigation

At perilune, the powered descent to the lunar surface will begin. This phase will begin at 9 NM (55,000 feet) altitude and continue until the space tug has soft-landed on the lunar surface. The tug may be manually controlled during this phase. There are three subphases associated with this maneuver which are: braking phase – deorbit from 9 x 60 NM orbit to lunar intersect trajectory; approach phase – approach to landing site using throttleable engine; landing phase – final approach and landing.

- Compute velocity and position with a sufficient accuracy to land the tug within a radius of 1.5 NM of the landing site with a landing velocity less than or equal to 1.5 ft/sec. The tug state vector will be updated using a landing radar and IMU accelerometer measurements as inputs to a Kalman filter navigation update scheme.

- Maintain attitude reference to an accuracy of ± 10 minutes of arc.

Guidance

- Issue commands to properly initiate and terminate main engine burn for descent to lunar surface.

- Provide attitude control commands to control the tug's engine thrust during the burn and to control vehicle attitude during coast phases while descending to the lunar surface.

- Provide engine throttle commands as required to properly control the tug's trajectory to the desired lunar landing site.

- Accomplish one of the following:

  (1) Guide the tug to a lunar landing with a guidance error less than 0.1 NM radius of a predetermined site for landing without a beacon.

  (2) Guide the tug to a lunar landing with a guidance error of less than 0.01 NM radius of a predetermined site for landing with a beacon.

- Guide the tug to the desired lunar landing site such that the tug's velocity relative to the moon is zero at an altitude of 5.0 feet.

Control

- Thrust vector alignment to the engine reference line shall be within TBD degrees (3σ; tentatively $\pm 0.5^{\circ}$).
- At engine start, thrust level gimbal deflection uncertainty shall not exceed TBD degree (3σ; tentatively $\pm 0.1^{\circ}$).
- Thrust vector angular deviation from zero thrust position due to gimbal deflection shall not exceed TBD degrees (3σ; tentatively $\pm 0.75^{\circ}$) when engine is operating at full throttle.
- Pitch, yaw, and roll attitude control shall be accomplished using the RCS and/or TVC.
- During manual operation, the crew shall be provided with attitude rate command capability. The commanded rates will be summed with the vehicle rate signal, and the attitude error input will be disconnected from guidance and shorted to low side externally. When the vector sum of the attitude rates about all axes goes below TBD (tentatively 1.0°/sec), the vehicle attitude at that time shall be held by means of RCS thruster commands.
- Response time to crew commanded rates shall be less than

$$\Delta\theta_2/a_2 + \text{TBD (tentatively 0.2 sec)}$$

$\Delta\theta_2$ = absolute value of the difference between the constant command rate and the initial vehicle rate.

a_2 = ratio of RCS torque to rigid body inertia.

- The final vehicle rate shall be within TBD (tentatively $\pm 0.4^{\circ}$/sec) of commanded rate.
- When the manual controller is dropped into detent, the maximum time for the vector sum of the attitude rates about all axes to go below TBD (tentatively 1.0°/sec) shall not exceed:

$$\frac{\Delta e_1}{a_1} + \text{TBD (tentatively 0.2 sec)}$$

Δe_1 = difference in two above rates

a_1 = ratio of RCS engine torque to rigid body inertia.

- Attitude rates from manual rate commands shall range nominally from +TBD (tentatively $+20^{\circ}$/sec) to -TBD (tentatively -20°/sec) about each axis.
- The main engine throttling commands shall be such that the vehicle rate of descent along the calculated local vertical is maintained at the desired rate.

- Incremental changes in the rate of descent shall be commanded by means of discretes, each discrete representing an incremental change of TBD (tentatively 1.0) ft/sec.

- Discretes shall be available to command positive or negative acceleration along each axis via the RCS.

(6) Deactivation Phase

There are no NG&C functions during this phase except those associated with checkout.

(7) Activation Phase

Navigation

- Alignment of the IMU will be accomplished by aligning the longitudinal axis to the lunar local gravity using IMU accelerometer outputs. Azimuth alignment is determined by measuring the angular relationship of the IMU yaw axis with respect to two cataloged celestial stars. The azimuth of the space stations orbit with respect to the landing site at the proposed liftoff time is calculated with reference to the same celestial stars and transmitted to the space tug on the lunar surface. The space tug then commands the yaw axis into the space station orbit plane in the downrange direction.

- The space tug state vector is updated while on the lunar surface by one of the following means:

  - The position and velocity of the space tug at the time of liftoff is transmitted to the space tug from either the space station or mission control on the earth.

  - The space tug receives signals from orbiting space station or earth and computes its astronomical position from this data.

  - The space tug calculates its astronomical position using the guidance computer whose inputs are the known lunar landing site location and prestored ephemeris table locations of the moon with respect to the selected inertial references.

Guidance

There are no guidance requirements for this phase.

Control

There are no control requirements for this phase.

(8) Powered Ascent Phase

Navigation

A 6 to 10 minute burn of the space tug will place the tug in a 9 NM by 45 NM elliptical orbit. There are two subphases associated with the powered ascent burn which are: vertical ascent and azimuth rotation to place yaw axis down range and guidance commanded pitch-over to attain desired orbit.

- Measure ascent burn velocity and position accuracy to ± 4 m/sec and ± 4 km, respectively.

- Maintain attitude reference accuracy to ± 10 minutes of arc.

Guidance

- Issue commands to properly initiate and terminate main engine burn for ascent from the lunar surface.

- Provide attitude control commands and engine throttle commands to control the tug's engine thrust during the ascent burn to lunar orbit.

- Guide the tug to a 9 x 45 NM orbit with apolune and perilune varying by no more than ± two kilometers from the above specified values.

Control

Control system requirements are same as those for the powered descent phase.

(9) Orbital Coast Phase

Navigation

The space tug will coast to apolune where a coelliptic circularization burn will be initiated. During this phase, the onboard systems will be monitored and configured for the next burn.

- Perform navigation update

- Align IMU

Guidance

- Provide attitude control commands to:

  (1) Control the tug's attitude during the coast phase.

  (2) Orient the tug and maintain attitude hold for engine firing to circularize at 45 NM.

Control

Control requirements are same as for the first coast phase.

(10) Orbit Circularization and Plane Change

Navigation

When the tug has reached apolune of the elliptical ascent orbit, the tug will be placed into a coelliptic 45 NM circular orbit.

- Measure coelliptic circularization burn parameters.
- Maintain attitude reference to an accuracy of ± 10 arc minutes.

The space tug will coast in this circular orbit until the nodal intersection between the plane of the space station orbit and the plane of the tug orbit is reached. A plane change maneuver, if required, will then be performed.

- Perform navigation update
- Align IMU

At the node between the space station and tug orbits a tug burn will correct for any inclination between the two orbits.

- Measure out-of-plane burn parameters.
- Maintain attitude reference to an accuracy of ± 10 arc minutes.

The space tug will continue to coast until one-half orbit after the coelliptic circular burn, where the tug will burn into a constant delta height (CDH) orbit with respect to the space station orbit.

- Perform navigation update
- Align IMU

The space tug is placed into a constant delta height orbit with respect to the space station.

- Measure burn parameters.
- Maintain an attitude reference to an accuracy of ± 10 arc minutes.

The space tug will coast in the CDH orbit until the central angle between the space station and the tug is 170 to 190 degrees.

- Perform navigation update
- Align IMU

Guidance

Guidance requirements for the burn phases are the same as the first burn phase and listed in paragraph (3). The coast phases are the same as the first coast phase and listed in paragraph (4).

Control

Control requirements are the same as those listed in paragraph (4) for the burn phase. Coast phase requirements are the same as those listed in paragraph (3).

(11) Rendezvous Phase

Navigation

When the central angle between the tug and the space station is between 170 and 190 degrees, the tug will burn into a transfer orbit to rendezvous with the space station.

- Measure transfer burn.
- Maintain an attitude reference to an accuracy of ± 10 arc minutes.

The space tug will coast to the midpoint of the transfer orbit where a midcourse correction will be performed.

- Acquire space station with rendezvous radar and provide range, range rate, line-of-site angle, and line-of-site angle rate as follows:
  - Range accuracy ± 0.5% or 1.0 meter
  - Range rate accuracy ± 0.5% or 0.1 meter/sec
  - Line of site (LOS) accuracy ± 0.1^{o}
  - LOS rate accuracy ± 0.1^{o}/sec
- Align IMU.

At the midpoint of the transfer orbit, the transfer orbit will be corrected by a midcourse correction burn.

- Measure midcourse correction burn parameters.
- Maintain attitude reference to an accuracy of ± 10 arc minutes.

The space tug will coast until final phase rendezvous.

- Continuous update of tug state vector.
- Align IMU.

The space tug will brake into a station keeping altitude with respect to the space station. This phase consists of placing the tug at a range of 1000 meters from the space station with a relative velocity of less than 5 ft/sec.

- Measure braking maneuver burn.
- Update the tug state vector with respect to the space station using a rendezvous radar or laser.
- Maintain attitude control within ± 10 arc minutes.

Guidance

Gross Rendezvous:

- Initiate and perform guidance to accomplish a burn to circularize the tug in a 45 NM orbit and subsequently to initiate short burns and coast periods to maneuver the tug to a coelliptic orbit approximately 15 NM below the LOSS by:

  (a) Issuing commands to initiate and terminate main engine burns.

  (b) Providing attitude control commands to control the tug's engine thrust and attitude during burn and coast periods.

    (1) Guidance during coast periods will consist of maintaining either a locally or inertially referenced attitude as determined by a preprogrammed sequence.

    (2) Guidance during thrusting phases will be closed loop using appropriate guidance laws for circularization and phasing (assuming optimum launch windows). The guidance system will contain the capability to operate in an automatic mode or by manual crew control.

Terminal Rendezvous:

- This mission phase begins at some variable time after the circularization burn to place the tug 15 NM below the target vehicle and includes the necessary maneuvers to achieve an adequate relationship to begin docking.

  (a) Guidance will be closed loop using a given reference system and rendezvous radar as sensors for targeting updates.

  (b) A burn-coast-burn capability will be included to:

    (1) Begin the transfer.

    (2) Make two midcourse corrections.

    (3) Final circularization.

(c) The terminal rendezvous period will end when the separation distance between the tug and target vehicle is reduced to approximately 1 NM.

Control

Control requirements are the same as for the previous burn and coast phases.

(12) Docking Phase

Navigation

A short series of burns will place the tug in the proper attitude with the desired rate to automatically dock the tug with the space station.

- Measure range, range rate, and attitude with respect to the space station as follows:

  - Range accuracy ± 0.5% or 1 meter
  - Range rate accuracy ± 0.5% or 0.1 meter/sec
  - Angular position accuracy ± 4.0 degrees
  - Angular rate accuracy ± 0.5 deg/sec

Guidance

- Begin guiding the tug to the target at a distance of not less than 1 NM from the target. The docking procedure shall consist of two phases. The first phase shall consist of guiding the vehicle from a 1 NM range to within a distance of 100 feet at a relative velocity of less than 0.5 ft/sec. During this phase, range rate will be maintained as a function of range.

- The second phase will guide the tug to impact with the target vehicle such that at impact the range rate is less than 1 ft/sec, the lateral rate is less than 0.5 ft/sec, and the lateral displacement is less than 1 foot.

- Attitude commands will be issued such that at impact the alignment error is less than four degrees and the attitude rate relative to the target is less than 0.5 deg/sec in each axis (pitch, roll, and yaw).

Control

- The attitude and translational control shall be accomplished within the following constraints.

  (1) Maximum closing velocity – TBD (tentatively between 0.5 and 2.0) ft/sec.

  (2) Maximum lateral offset – TBD (tentatively between 5 and 18) inches.

(3) Maximum lateral rate – TBD (tentatively between 0.1 and 0.75) ft/sec.

(4) Maximum alignment angle – TBD (tentatively between 5 and 10) degrees.

(5) Maximum angular rate – TBD (tentatively between 0.25 and 1.0) degrees/sec.

- The attitude and translational control system shall respond to commands issued dependent upon relative attitude, position, and rates of change between the tug and the LOSS.
- The required commands shall be issued to the RCS.
- In contrast to undocking, the maneuver may be time critical and monitoring of precise attitude and translational control is a requirement.
- The attitude control subsystem deadband shall not exceed TBD (tentatively $\pm$ 0.5) degrees.

3.2.2 Supporting Subsystems Requirements

The navigation, guidance and control subsystem functional requirements and minimum performance requirements are mission and mission phase dependent. The functional requirements of several of the astrionic subsystems are not necessarily mission dependent but are mission phase (e.g., boost, coast, storage, etc.) dependent. These subsystems or functions include:

(a) Data Monitoring and Reporting

(b) Command

(c) Tracking

(d) Sequencing

(e) Power Generation

(f) Power and Signal Distribution

(g) Structures, Mounting and Packaging

(h) Checkout

3.2.2.1 Data Monitoring and Reporting

A data monitoring and reporting function shall be provided to permit the mission control personnel (crew or ground) to monitor and evaluate vehicle performance at major decision points in the mission. The data monitoring and reporting function shall be capable of monitoring all critical vehicle parameters and reporting and/or displaying the resulting data to the users.

The voice and telemetry transmission shall be compatible with all space resident vehicles, a data relay satellite system and the MSFN and Deep Space receiving and data processing networks with respect to frequency, data rates, and modulation techniques. Pulse Code Modulation (PCM) and/or Frequency Modulation (FM) shall be used for the operational measurements. The maximum PCM error rate shall be (TBD).

The vehicle transmitting antennas shall be located to provide adequate communications capability with a minimum of antenna pointing control. The communications equipment shall have a minimum transmitting distance of a few feet to a maximum distance from the earth to the moon. The transmitting antenna shall have a minimum beamwidth of (TBD).

The analog data accuracy shall be a minimum of ± 5% of the full scale measurement range. The repeatability, or maximum difference in calibration points found in a series of repeated calibrations under the same test conditions, shall not exceed 25 mvdc nominal.

3.2.2.2 Command

The command uplink subsystem is the onboard receiver(s) used for real-time command capability and/or remote vehicle control for various missions. The commands may be generated and/or initiated from ground control or from other space vehicles.

The requirements for the command function for the space tug are as follows:

(1) The command uplink function shall receive uplink data, demodulate/decode it, verify it, determine the system to which it is destined, and direct it to that system.

(2) The command uplink function shall have, as a minimum the capability to provide timing updates, navigation and guidance function updates, and switching commands to the tug data management system.

(3) The command uplink function shall have the capability to provide data to the crew (if manned) or the initiator of the command to monitor and evaluate vehicle reaction during the command execution process.

(4) The probability of the command uplink function to process erroneous or invalid commands shall be less than or equal to (TBD) for individual commands to the tug. This assumes valid and correct information is transmitted to the tug.

(5) The command function shall be independent of other space vehicles and shall be compatible with ground and space vehicle command stations.

3.2.2.3 Tracking

Tracking is the function of determining the vehicle rate of change of position and/or position by the use of a source external to the space vehicle. It also applies to onboard tracking aids used to determine other vehicle or object rate of change and/or position relative to the source vehicle.

Although autonomous operation of the tug is a desired design feature, external tracking of the space tug is required to determine the position and velocity of the tug by ground-based external tracking or space-based vehicle tracking. The ground-based tracking is required to aid the tug during unmanned missions and as a backup to the tug navigation system. Tracking by other space elements, such as the space station is required by the space vehicle to aid the tug in rendezvous and docking and for collision avoidances. The tracking aids on the tug must be compatible with space and ground-based tracking systems.

3.2.2.4 Sequencing

Sequencing is the process of determining the order and the time of commanding the execution of events (both internal and external to the sequencing subsystem), critical mission timing and the operation of some vehicle equipment.

The sequencing function shall be capable of meeting the following requirements:

(1) To sequence time-dependent events, a clock reference signal whose frequence instability does not exceed ± 2 parts per million shall be provided.

(2) The capability to accumulate, access, and display elapsed mission phase and/or subphase time shall exist.

(3) The capability to sequence events relative to the recognition of other mission critical events.

(4) The capability to recognize, determine and command events both internal and external to the astrionic system shall exist.

(5) Sequencing shall be performed consistent with the vehicle systems configurations and timing accuracy requirements.

(6) The capability shall exist to sequence manually, automatically or through the space tug command subsystem.

(7) The capability shall exist to sequence functions as required to assist the checkout and maintenance functions.

(8) The capability shall exist to vary the time for initiating certain operations (such as undocking).

3.2.2.5 Power Generation

The power generation subsystem shall furnish suitable electrical power for satisfactory operation of all or a portion of the various subsystems. A sufficient quantity of electrical energy shall be supplied to enable the satisfactory completion of the mission.

The requirements which follow are written without regard to mission phase.

- The power generation function shall contain a primary electrical power source which shall generate a suitable direct current voltage to power most or, if no special voltages are required, all astrionic system components/subsystems. Primary source capacity shall be sufficient to permit successful completion of all mission phases.

- The power generation function shall contain, if necessary, special electrical power devices which shall convert the power furnished by the primary source to a form required by certain components/subsystems.

- The power generation function shall exhibit the capability of detecting electrical faults, such as electrical shorts, and isolating these faults such that no excessive electrical transients are generated or without excessive drainage of primary electrical power.

- The primary power source shall be compatible with external power sources.

- The primary power source shall be capable of acting as a standby or backup power source whenever external power sources are used.

- The primary power source shall be capable of being operated in the following manner without excessive electrical transients being generated:

  (1) Switching the astrionics from external to internal power and back again.

  (2) When powering up or powering down the astrionics while on internal primary power.

- The power generation function shall contain a primary power source capable of being placed in a quiescent state, remaining in the quiescent state for up to 42 days, and then being reactivated to full operating condition. The quiescent state shall be characterized by a minimum rate of energy consumption.

- The primary power source shall be capable of being reactivated or replaced during scheduled maintenance phases.

3.2.2.6 Power and Signal Distribution

The power and signal distribution subsystem distributes electrical power to the various components comprising the astrionic system and routes signals between these electrical components.

The requirements that follow are written without regard to mission phase.

- This subsystem shall distribute electrical power to the various components comprising the astrionic system.

- This subsystem shall route analog, digital and/or discrete electrical signals between components comprising the astrionic system.

- To facilitate maintenance and checkout, flexibility shall be provided through the use of electrical distributors, as required.

- This subsystem shall provide switching, both momentary and latching, to facilitate sequencing and signal substitution for redundant functions, checkout, and powering up and powering down of components.

- Provision shall be made to switch between external and internal power.

- This subsystem shall provide a means by which the onboard astrionics can interface with external systems for monitoring, checkout, and data input/output functions. Such astrionics umbilical connections are required for:

  (1) Launch pad operations prior to launch.

  (2) Space Station operations while in orbit.

  (3) RNS operations while the tug and RNS are docked.

3.2.2.7 Structures, Mounting and Packaging

This subsystem includes all secondary structures, component mounting bracketry and component packaging required to install the astrionic subsystems in the space tug.

(1) Primary Structures – Defined as the major load bearing shell and framework of the stage.

(2) Secondary Structures – Defined as astrionic equipment mounting panels, plates or stringers which bolt, rivet, bond, etc., to the primary structure.

(3) Component Mounting Bracketry – Defined as the brackets or mounts used to attach the individual component to the secondary structure.

(4) Component Packaging – Defined as the exterior housing or container used to protect or contain the astrionic hardware.

The primary structure must provide its load carrying function plus the following additional functions and/or conditions:

- Loads environments

- Equipment mounting

- Accessibility

- Environmental protection

- Modular interface

- Weight

Preliminary requirements for the primary structure are given in Appendix I of this report.

The secondary structures, component packaging, and component mounting bracketry shall be designed to a TBD (tentatively 1.1) yield factor of safety, or a TBD (tentatively 1.4) ultimate factor of safety, whichever governs.

The only phase dependent requirements are those associated with the maintenance phases of the mission which contain maintenance requirements.

- Access shall be provided to the astrionics to facilitate servicing and maintenance of astrionic equipment.

- Maintenance actions shall be remove-and-replace operations. These operations shall be conducted in a TBD environment. (Possible environments are pressurized environments in which space suits would not be required or unpressurized environments in which space suits would be required.)

- Astrionic system layout shall, as a goal, minimize maintenance time. Special attention shall be directed toward reducing to a minimum the number of components resulting in maintenance actions which will consume more than TBD hours of elapsed time from malfunction detection through maintenance and checkout.

- Astrionic component layout shall be such that (1) components are easily accessible and (2) the remove-and-replace operations will cause minimum disturbance to adjacent equipment.

- Astrionic component mounting methods shall be designed to minimize the number of special tools and/or procedures required for the remove-and-replace operation.

- Component packaging envelope configurations shall meet the following requirements:

  (1) Equipment shall be packaged in units that will not exceed TBD (tentatively 20) inches x TBD (tentatively 25) inches x TBD (tentatively 40) inches.

  (2) Friction hinges shall be utilized so that hinged devices will remain as positioned by the crew.

  (3) Each transferrable component shall have a minimum of one hand hold and one tether attach point. The attach points shall be in as close proximity to the center of the container as possible and in line with the center of mass in the direction of transfer.

  (4) Forces for manual release systems shall not exceed TBD (tentatively 25) pounds lateral force or TBD (tentatively 45) pounds pull.

  (5) For purposes of identification, components shall be marked and/or color coded.

3.2.2.8 Checkout

Checkout includes the process of determining the operational capability of the vehicle, diagnosing malfunctions, and reverification of operational status after a repair sequence. Checkout, as discussed here, includes pre-flight or pre-mission checkout, and operational checkout as required in various mission phases.

A basic design goal for the space tug is autonomous operation. The autonomous operation includes checkout capabilities. Therefore, a design goal for the checkout system is the capability of checkout of the system independent of external controls or monitoring equipment.

The following terms are defined to indicate the effects of certain item failures on the mission and to establish a baseline for capability of the checkout system:

(1) Criticality I failures – Jeopardize the safety of the crew.

(2) Criticality II failures – Cause primary mission abort.

(3) Criticality III failures – Cause reduction in fulfilling mission objectives.

The following requirements are independent of mission phase:

(1) Detection of a critical malfunction shall result in switching to an alternate unit, module, path, or method to complete the mission successfully. Crew and Mission Control alert shall be provided.

(2) Monitoring parameters for trends and determining a prediction of the time that an unsatisfactory condition will exist shall be considered for incorporation. The capability shall exist to alert the crew and Mission Control of equipment degradation or potential equipment failures.

(3) A preset testing sequence using operational inputs as stimuli and analyzing outputs shall be provided.

(4) A preset testing sequence using test stimuli, either generated internally or externally, to the unit and analyzing outputs shall be provided.

(5) Programs necessary for diagnostic testing using data obtained from a number of sources shall be provided. Diagnostic testing is required when the cause of a problem is not readily apparent from the individual monitor points. Diagnostic testing consists of analyzing symptoms or trends from interrelated test results and determining the nature of the cause. Fault isolation to the lowest replaceable unit (LRU) is a goal.

(6) The checkout function shall provide a capability of checking the entire tug (astrionics plus other modules and payload systems) by monitoring parameters and comparing the values against preset limits or expected sequence of events.

(7) The checkout function shall be capable of remote activation and monitoring.

(8) The checkout function shall be so designed as to not adversely affect the component or system under test.

(9) Onboard vehicle checkout equipment and techniques shall be compatible with existing launch and mission support facilities.

(10) The checkout function shall be designed to detect all failures where practical. Criticality I and II items shall be failure detectable to the maximum extent possible.

The following paragraphs contain the checkout functional requirements peculiar to each phase of the mission:

Reactivation or Power-up – the checkout function must perform a complete checkout of the space tug subsystems. Testing will include complete end-to-end testing for all subsystems. Redundant, backup, and alternate paths shall be tested when practical.

Flight Phases – The checkout function will monitor operational parameters necessary to determine the operational status of the space tug. Limit checks, reasonableness tests, trend analysis, and detection of built-in alarm indications will be utilized. End-to-end testing of propulsion subsystems shall occur prior to engine burns. Emphasis of the testing will be on navigation, tracking and propulsion equipment/subsystems.

Deactivation – Prior to deactivation, a complete checkout as described earlier will be performed. Any anomalies noted during previous phases will be investigated and resolved. The checkout function shall then be deactivated except for that portion required to support any system functions still active.

3.3 ASTRIONIC SYSTEM/TUG OPERATIONS INTERFACE

The interface of astrionic system functions in overall space tug operations is assessed relative to a functional flow diagram.

A top level Functional Flow Diagram for typical tug operations is shown in Figure 3-2. The major system elements included in this functional flow in addition to the space tug are:

(a) Transport System – Vehicles for transporting the space tug to its operational station may be an EOS or a Saturn V derivative for transport to LEO. For the LPO operations, the transport system may be an EOS and RNS or a four stage Saturn V vehicle.

(b) Operational Base – The operational base provides necessary assistance and logistic support for the prepare-for-mission operations and includes such functions as maintenance, fueling, and reconfiguration of the tug, as required, for the assigned mission.

(c) Mission Control – A monitor and control function is required to schedule and monitor the tug missions. Mission Control may be at an earth base or at a space station. For some missions navigation data will be required.

The following paragraphs relate the astrionic system preliminary functional requirements to tug synchronous orbit mission operations. These functional requirements are presented in chronological order within the functional flow diagram (Figure 3-2) blocks as follows:

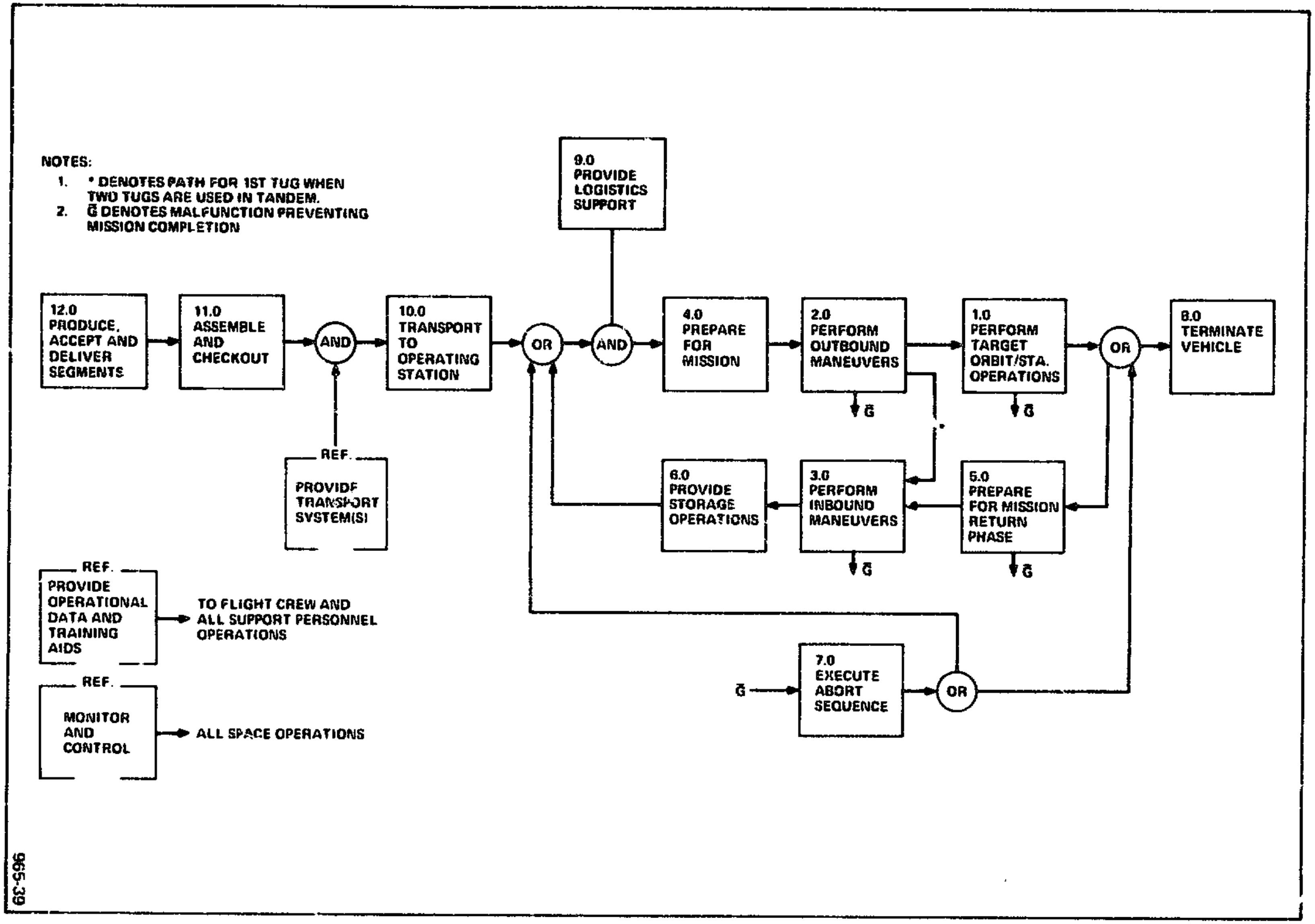

Figure 3-2. Top Level Space Tug Functional Flow Diagram

(a) Prepare for Mission (tug function 4)

(b) Perform Outbound Maneuvers (tug function 2)

(c) Perform Target Orbit/Station Operations (tug function 1)

(d) Prepare for Return Mission Phase (tug function 5)

(e) Perform Inbound Maneuvers (tug function 3)

(f) Provide Storage Operations (tug function 6)

Follow-on work, dependent on more definition of tug vehicle and its operations, is needed to fully relate astrionic system requirements to tug operations.

The unmanned synchronous orbit mission, as illustrated in Figure 3-3, transfers a payload from LEO to equatorial synchronous orbit, rendezvous and docks to a satellite, performs on-station operations, returns to LEO, and rendezvous and docks to the space station. Two tugs are used in tandem in the initial mission phase. The first tug, after the initial transfer burn, returns to the space station.

3.3.1 Prepare for Mission (4.0*)

The objective of this function is to activate and checkout the tugs, perform any necessary maintenance, perform servicing and fueling operations, and assemble vehicle and payload. Included are provisions for all orbital support operations. This function starts with tug activation and terminates with verification of vehicles flight readiness.

A. DESIGN CHARACTERISTICS

(1) The astrionic system shall be capable of being powered up from a storage and/or quiescent state to an operational status within two hours.

(2) A tug checkout, including astrionics, shall be performed. Automatic onboard checkout equipment will be utilized to the maximum extent.

(3) Tug fueling operations will be monitored and controlled by the astrionic system.

(4) Tug translation and attitude positioning/hold requirements during vehicle and payload assembly shall be TBD.

(5) The astrionic system digital computer must be capable of being loaded (and verified) with a new flight program and/or specific navigational data and operating instructions, as required, for assigned missions.

*Function number per FFD (Figure 3-2).

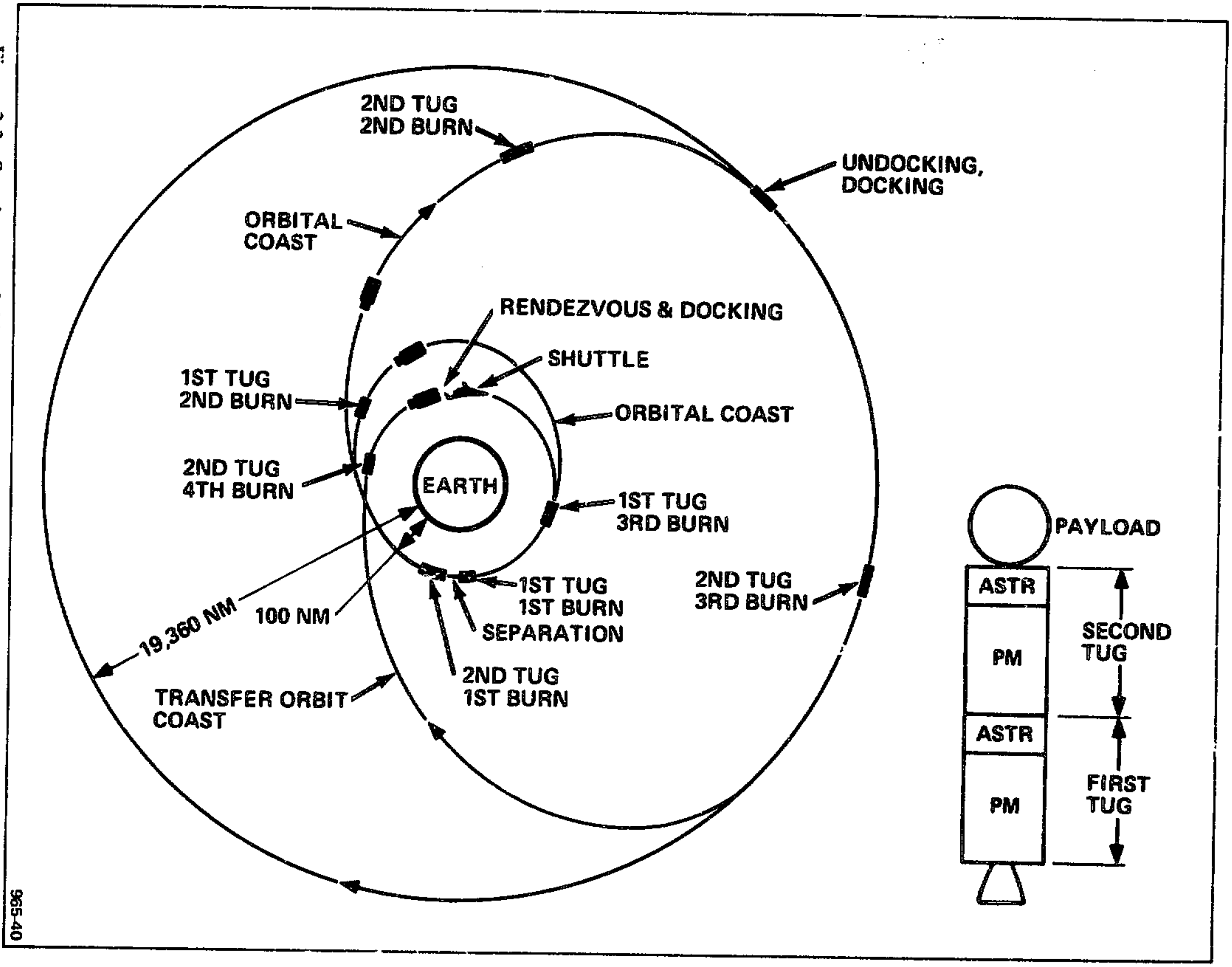

Figure 3-3. Synchronous Orbit Mission Profile and Vehicle Configuration (Reusable Stages)

(6) Just prior to performing outbound maneuvers (2.0), the onboard NG&C system shall:

(a) Initialize the navigation system with tug state vector received from space station. Accuracy of state vector initialization shall not be less than

- Position – TBD
- Velocity – TBD
- Time – TBD

(b) Align the platform. Accuracy of alignment shall be as required to provide not less than TBD minutes of arc in all three axes during the transfer burn.

B. DESIGN CONSTRAINTS

(1) The natural environments encountered are as defined in NASA TMX-53798. The induced environments are those encountered during translation and attitude maneuvers, rendezvous and docking, vehicle assembly, and fueling operations.

(2) Time allocated for tug checkout shall not exceed TBD minutes.

C. EFFECTIVENESS

(1) Reliability - After completion of checkout and any needed maintenance, the astrionic system shall not contribute more than TBD to the tug unreliability in performance of the assigned mission.

(2) Safety – TBD

(3) Maintainability – Provisions shall be made for access to the astrionic equipment, and replacement to the lowest replaceable unit shall be accommodated.

D. INTERFACE

(1) When docked to the space station/base, interface with the space station (hardwire) shall be provided to permit:

(a) Manual initiation and monitor of tug checkout.

(b) Loading new flight program and/or special operating instructions and/or navigational data/parameters.

(c) Interchange of power and thermal conditioning.

(2) When free-flying near the space station/base, interface(s) with the space station (RF links) shall be provided to permit:

(a) Receiving special discrete commands/instructions and navigational data/parameters.

(b) Transmission of tug status data to the space station.

(3) Interfaces with payloads shall be as defined in applicable Interface Control Documents.

(4) Interfaces with another tug when connected in tandem.

3.3.2 Perform Outbound Maneuvers (2.0)

The objective of this function is to perform those operations required to deliver the payload into the required synchronous orbit position. Included in this phase are the prescribed boost-coast phases for transfer. This function is initiated on completion of all vehicle assembly and checkout operations and terminates on achieving synchronous orbit. This function includes the following subfunctions.

(a) Orbital Maneuvers and Coast – Maneuver the vehicle a safe distance from the space station and coast until the required position (at equator and the line of nodes in the vicinity of the target orbit longitude) is reached.

(b) Execute Transfer Burn and Staging – The first tug will provide initial transfer velocity and then separate. The second tug will complete the transfer burn.

(c) Coast – Coast until synchronous altitude is reached (second tug with payload).

(d) Circularization and plane change burn – Main engine burn to execute plane change and circularize orbit. (Second tug).

A. DESIGN CHARACTERISTICS

(1) The NG&C subsystem shall provide vehicle/engine controls, as required, to provide a flight trajectory resulting in achieving an equatorial synchronous orbit having the following nominal characteristics:

Nominal Characteristics: TBD

Note: NG&C accuracy requirements are dependent on degree of autonomy required and trajectory flown. Navigation analysis has defined four potential trajectories (direct ascent and slow drift phasing, low phasing orbit, high phasing orbit, and overshoot phasing). Nominal NG&C accuracies were defined for three of these trajectories based on the requirement of being within radar range on achieving synchronous orbit altitude. In-depth analysis is required to select optimum trajectory and determine degree of autonomy available and to provide an error budget.

(2) During coast periods, vehicle attitude position(s) shall be provided as required to meet solar heating and/or antenna pointing requirements.

(3) Internal status data is to be provided to mission control.

(4) Provide cooperative tracking for ground stations.

(5) Receive and execute any targeting or navigation data updates.

B. DESIGN CONSTRAINTS

(1) Nominal time for this function is TBD hours.

(2) Off-set targeting will be used to minimize total energy required and to place tug in near optimum position for the rendezvous phase.

(3) Environments:

Natural environments – (same as Function 4.0; see Sec. 3.3.1)

Induced environments – TBD

(4) The astrionic module is to be self-sufficient.

(5) In event of mission abort – TBD.

C. EFFECTIVENESS REQUIREMENTS

(1) Reliability – The astrionic module shall contribute no more than TBD to the tug mission unreliability.

(2) Safety – TBD

(3) Maintainability – In-flight maintenance shall be limited to automatic switching of redundant components or subsystems.

D. INTERFACES

Communication links are to be provided to:

(a) Receive discrete commands/instructions and navigational parameters/data from mission control.

(b) Transmit status data to mission control.

3.3.3 Perform Target Orbit/Station Operations (1.0)

The objective of this function is to rendezvous and dock with a cooperative satellite and transfer payloads. This function is initiated on completion of the plane change and orbit circularization maneuvers and terminated on completion of the on-station operations. This function includes the following subfunctions:

Orbital Coast/Phasing – Vernier burns may be required to achieve position within radar range of target satellite.

Rendezvous Maneuvers – Series of vernier burns to place tug in position for docking to satellite.

Dock – Series of vernier burns to achieve docking to satellite.

Station Keeping - Provide and hold attitude position(s) for payload transfer.

Undock -- Vernier burn to separate tug from satellite.

A. DESIGN CHARACTERISTICS

(1) The astrionic system shall contain the necessary onboard facilities to locate and track target satellites and compute and execute trajectory/maneuvers for rendezvous. At completion of this rendezvous the tug shall be in a TBD position (about 1 NM below and behind target) for initiation of the docking phase.

(2) The astrionic system shall provide the necessary [illegible]C to execute the docking maneuvers.

Note: Docking tolerances are dependent on docking mechanism used.

(3) Onboard sensor(s) and communications link will be provided to permit visual monitoring of docking and payload transfer operations by mission control.

(4) The astrionic system shall receive and execute discrete maneuver orders from mission control to provide vernier payload positioning in orbit.

(5) Internal status data is to be provided to mission control.

(6) Provide cooperative tracking for ground stations.

(7) Receive and execute any targeting or navigation data updates.

B. DESIGN CONSTRAINTS

(1) Docking target shall be cooperative.

(2) Nominal time for this function is TBD hours.

C. EFFECTIVENESS REQUIREMENTS

Same as for Function 2.0, see Sec. 3.3.2.

D. INTERFACES

(1) Communicative link required for visual monitor transmission to mission control.

(2) Target satellite will contain TBD system to facilitate location and tracking for the rendezvous and docking operations.

(3) Same as Function 2.0, see Sec. 3.3.2.

3.3.4 Prepare for Return Mission Phase (5.0)

This function is to provide any navigation and/or target updating and orbital phasing needed prior to initiation of return transfer burn.

A. DESIGN CHARACTERISTICS

TBD

B. DESIGN CONSTRAINTS

Same as Function 2.0, see Sec. 3.3.2.

C. EFFECTIVENESS

Same as Function 2.0, see Sec. 3.3.2.

D. INTERFACES

Same as Function 2.0, see Sec. 3.3.2.

3.3.5 Perform Inbound Maneuvers (3.0)

The objective of this function is to perform those operations required to transfer the tug and payload from synchronous orbit to an orbit near the space station. This function includes the following subfunctions:

- Plane change and brake into transfer ellipse.
- Coast in transfer ellipse perigee.
- Brake into near circular orbit.
- Rendezvous and dock to space station.

A. DESIGN CHARACTERISTICS

(1) The NG&C subsystem shall provide vehicle/engine controls, as required, to provide a flight trajectory resulting in achieving a target in low earth orbit having the following nominal characteristics:

Target Orbit Characteristics: TBD

(2) During coast periods, vehicle attitude position(s) shall be provided as required to meet solar heating and/or antenna pointing requirements.

(3) Internal status data is to be provided to mission control.

(4) Provide cooperative tracking for ground stations.

(5) Receive and execute any targeting or navigation data updates.

(6) The astrionic system shall contain the necessary onboard facilities to locate and track the space station and compute and execute trajectory/maneuvers for rendezvous. At completion of this rendezvous the tug shall be in a TBD position for initiation of the docking phase.

(7) The astrionic system shall provide the necessary NG&C to execute the docking maneuvers.

Note: Docking tolerances are dependent on docking mechanism used.

(8) Onboard sensor(s) and communications link will be provided to permit visual monitoring of docking operations by mission control.

(9) The astrionic system shall receive and execute discrete maneuver orders from either mission control or the space station.

B. DESIGN CONSTRAINTS

(1) Nominal time for this function is TBD hours.

(2) Environments:

Natural environments – (same as Function 4: See Section 3.3.1)

Induced environments – TBD

(3) The astrionic module is to be self-sufficient.

(4) In event of mission abort – TBD.

C. EFFECTIVENESS

Same as Function 2.0: see Sec. 3.3.2.

D. INTERFACES

(1) Communication links are to be provided to:

(a) Receive discrete commands/instructions and navigational parameters/data from mission control.

(b) Transmit status data to mission control.

(c) Transmit visual monitor data (TV) to mission control.

(2) When docked to the space station/base, interface with the space station (hardwire) shall be provided to permit:

(a) Manual initiation and monitor of tug checkout.

(b) Interchange of power and thermal conditioning.

3.3.6 Prepare for Storage (6.0)

The objective of this function is to power-down the astrionic system for storage.

A. DESIGN CHARACTERISTICS

During the stored or quiescent period periodic status data is to be provided to mission control or space station on command.

B. DESIGN CONSTRAINTS

Storage periods of up to 180 days, either docked to a space station or free-flying, are required.

C. EFFECTIVENESS

TBD

D. INTERFACE

(1) When docked to the space station, interface with the space station as specified in Function 4.0 (see Sec. 3.3.1) plus providing a power-up/power-down control to the space station.

(2) When free-flying, an RF link is to be provided to receive and execute discrete orders from the space station, including power-up/down, initiate self checkout, and transmit status data.

After receiving and executing a power-up order, station keeping orders/data are to be received and executed.

APPENDIX C

SPACE TUG DATA MANAGEMENT
ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

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1.0 INTRODUCTION

The purpose of this appendix is to describe the data management function, its requirements and the associated hardware required for the various space tug missions which are:

- Synchronous Orbit (expendable vehicle)
- Synchronous Orbit (reusable vehicle)
- Lunar Landing
- Earth Orbital Operations
- Lunar Orbital Operations
- Unmanned Planetary
- Reusable Nuclear Shuttle (RNS)
- Four Stage Saturn V

Functionally, data management integrates the tug astrionic system by providing for all subsystem-to-subsystem signal flow. In addition, data management provides the computational support for the following astrionic functions:

- Attitude control (TVC and RCS)
- Engine control
- Guidance
- Navigation
- Checkout and hardware reconfiguration control
- Data monitoring and telemetry
- Sequencing
- Data bus control
- Software management
- Display support
- Strapdown IMU processing
- Maintenance support
- Refueling support

Specifically, the following tasks were addressed in detail:

(1) Identification of constituent data management elements and the presentation of the functional operation of each, including trade-off considerations where applicable.

(2) Establishment of data management organization, subsystem-to-subsystem interface approach, and redundancy approach.

(3) Establishment of software organization and control.

(4) Determination of memory storage and CPU speed requirements.

2.0 STUDY GUIDELINES AND GROUNDRULES

The synchronous orbit mission was used as a baseline for the study. Requirements (hardware and software) were added or deleted to arrive at total requirements for the other missions.

The following groundrules were used as guidelines on which this study was based:

- Design will minimize the necessity for ground support during flight; autonomy is the design objective. Other design objectives are (1) minimize total weight, power consumption and volume, (2) minimize impact of adding or deleting component or subsystems to accommodate the various missions, (3) minimize impact of software modifications by minimizing the required verification effort while maintaining assurance in sufficiency of effort, and (4) maximize system reliability, crew safety and probability of fulfilling all objectives.

- Maintenance in space is required.

- The astrionic system must operate automatically and be capable of receiving remote commands from the earth, moon or other spacecraft.

- The astrionic system must be capable of maintaining a quiescent status for up to 180 days in earth or lunar orbit, both docked to other vehicles and free flying.

- The astrionic system must be capable of going from the quiescent state to a fully operational state within two hours.

- Critical systems which are necessary for crew survival will be designed such that no single failure or credible combination of failures will result in loss of life.

- The astrionic system will interface with the crew module to provide display of onboard status. The crew will be capable of manual control and operation of the data management subsystem.

- The astrionic system will be capable of operation without maintenance or refurbishment during operational periods and any time it is not docked to a space station.

3.0 SUMMARY

Based on current requirements for the various space tug missions, the data management hardware will consist of:

(1) A central processing unit (CPU) of medium speed range (300K to 400K ops/sec.)

(2) 32K to 64K of random access main memory

(3) A bus control and input/output unit

(4) A data bus system with standard interface units which interface all astrionic subsystems to the data bus

(5) A configuration assignment unit (CAU) which, in addition to monitoring itself, monitors the main memory, the CPU and bus control unit, and switches to backup units when a primary unit fails (see ref. C-7).

(6) Primary and auxiliary measurement units for sampling command, propulsion and astrionic module parameters

(7) A magnetic tape unit for mass storage

(8) A 32K display memory for storing display skeletons

(9) An auxiliary monitor computer.

Figure 4-1 shows a basic functional layout for space tug data management. Since component redundancy varies over the space tug mission spectrum, no attempt was made to depict redundancy in this figure. However, a backup CPU is shown to better illustrate the configuration assignment unit linkage with other devices.

The auxiliary monitor computer and the auxiliary measurement units are used only for the lunar landing mission to allow the primary system to be powered down during the 28 or 42 day period the space tug is on the lunar surface. By having the primary data management equipment powered down during this period, the overall reliability is maintained at a high level.

Critical components such as the CPU, data bus, some standard interface units (SIU), configuration assignment unit (CAU) and the bus control and I/O unit will have one backup unit each for unmanned missions and two backup units for manned missions with error detection and switching of the above items being provided by the CAU. This provides fail/safe operation in the first case and fail/operate-fail/safe operation in the latter case. Error detection for data and addresses on the data bus will be provided by encoding all data and addresses.

SIU error detection will probably be provided by SIU self detection hardware; and errors in sensor hardware will be handled by software (reasonableness test, limit test, and trend analysis).

The data bus concept was chosen for subsystem-to-subsystem communication in the astrionic system for the following reasons:

(1) The data bus has as a feature a standard digital interface that can be specified for components performing varied functions in large and complex astrionic systems.

(2) The data bus concept offers the advantage of flexibility in that subsystems or components can be added, deleted, or upgraded with little or no effect on the system interface.

(3) The data bus has a weight advantage over the conventional centralized system because data bus multiplexing allows all data to and from subsystems to be handled by one or a small number of cables. This supports the space tug objective to minimize weight.

(4) The data bus realizes a reliability advantage through a more efficient system functional design and because interface circuitry, wiring, and connectors are reduced.

(5) Maintenance is simplified by eliminating signal distributors and multiple connectors, by standardizing a communication technique and by providing ready access to components that need to be added or deleted without impacting the entire system.

Results show that use of a conglomerate software organization for space tug is not tenable; thus, modular organization will be used for the data management main memory software. Using this approach, an executive control program will control all CPU operations. The software design goal is to totally isolate each function module so that a program change in any module will not affect other modules. Thus, for a given phase of the mission, executive control program will load from bulk storage into main memory only the required function modules.

A preliminary estimate was made to determine worst case CPU speed and random access memory requirements. These estimates show a worst case need for 62,760 32-bit memory locations and a CPU speed capability of 278,000 operations/second. However, the requirements for supporting the refueling operation has not been assessed due to lack of information. Indications are that the CPU speed requirement for refueling may exceed the 278,000 ops/sec. required during the worst case mission. This problem will be addressed as information becomes available. In addition, CPU speed and memory requirements for RNS nuclear engine control were not included due to the lack of information and complexity of this function. Preliminary information indicates requirements for this function could easily double the mission requirements for RNS shown in Table 4-1.

4.0 DETAILED ANALYSIS

4.1 DESCRIPTION

Data management provides the control for all data and signal flow throughout the space tug. Figure 4-1 depicts the basic data management functional block diagram While the basic astrionic components for all missions are included in the diagram, component redundancy is not because it is not the same for unmanned missions as it is for manned missions. As noted on the diagram, the displays and associated hardware are for manned missions only; in addition, the auxiliary computer and its associated monitoring equipment are included in the lunar landing mission only. This hardware is included to monitor critical measurements and for display support during the 28 or 42 day period the space tug is on the moon; otherwise, the main computer and data bus system would have to be powered up during this period, in which case reliability would be significantly reduced.

Determination of system characteristics, organization, redundancy, functional operation, and interface with other subsystems are the major topics of this discussion. The following paragraphs detail these topics.

4.1.1 Random Access Memory (RAM)

The quantities of random access memory (RAM) required to accommodate data management functional requirements for each space tug mission are presented in Table 4-1. Based on this table, each mission will have either 32K or 64K of RAM.

A study was made to determine what memory technologies should be considered for use in implementing the required RAM. The currently available RAM technologies are as follows (see reference C-1):

- Magnetic core
- Magnetic film memories
  - (1) Plated wire (PW)
  - (2) Planar magnetic film (PMF)
- Metal oxide silicon (MOS) } Monolithic technology
- Bi-polar devices } Monolithic technology

The following paragraphs discuss the more important characteristics of these technologies

4.1.1.1 Memory Speed

Figure 4-2 shows projected 1972 read-write cycle times for the above mentioned technologies. Preliminary estimates indicate cycle times of one μ sec or less will be required to adequately perform the required functions. Thus, non-destructive readout (NDRO) core is not a candidate for space tug application from an access speed point of view. That is not to imply that NDRO is undesirable or unavailable in monolithic technology in the near future in LSI devices.

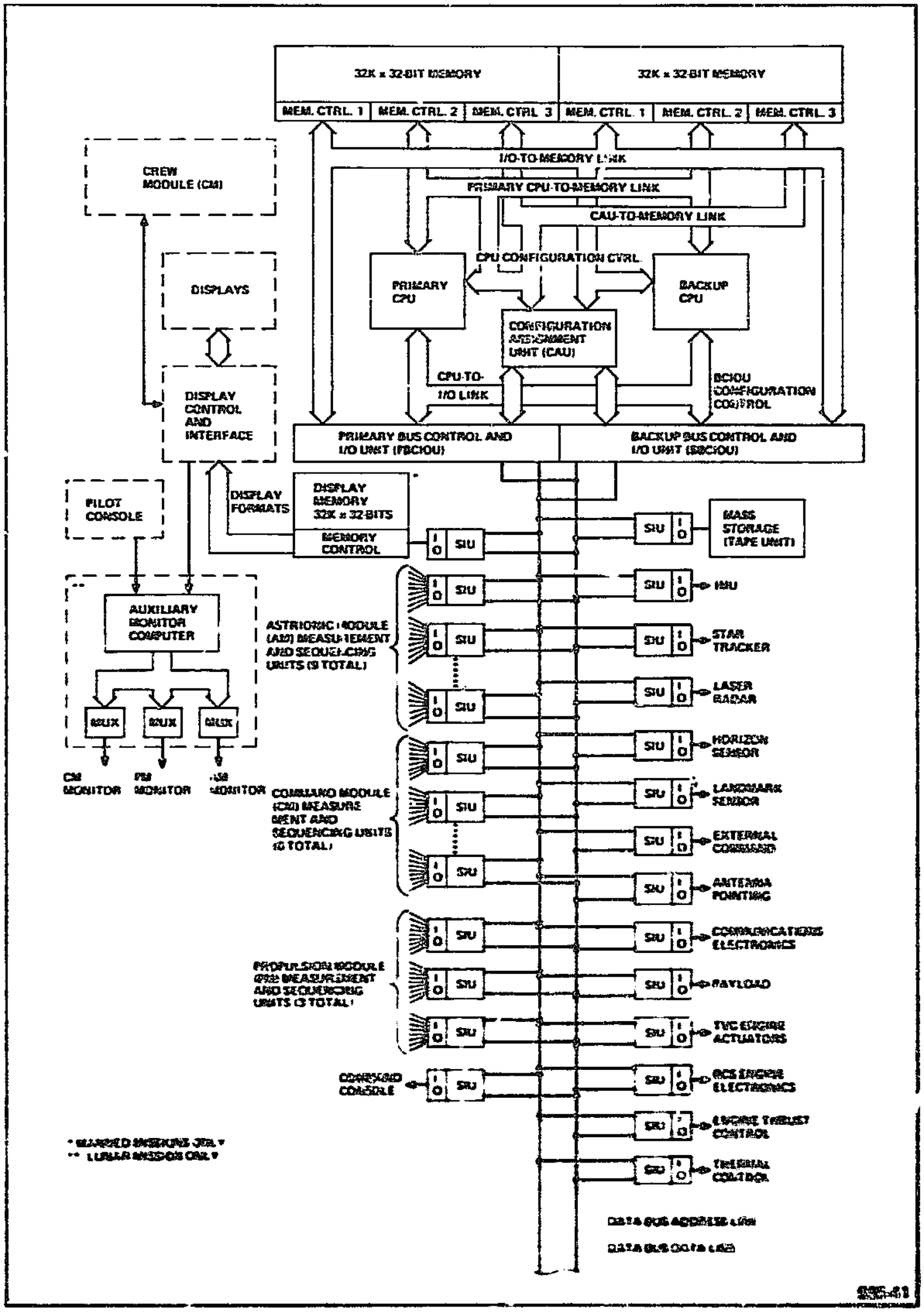

Figure 4-1 Space Tug Data Management Functional Layout

Table 4-1. Memory and CPU Requirements

[illegible]	Sync. Orbit [illegible] Memory 32 Bits	Sync. Orbit [illegible] Ops/Sec*	Sync. Orbit + Reusable Veh. Memory 32 Bits	Sync. Orbit + Reusable Veh. Ops/Sec*	Sync. Orbit [illegible] Reusable Veh. Memory 32 Bits	Sync. Orbit [illegible] Reusable Veh. Ops/Sec*	Lunar Landing Memory 32 Bits	Lunar Landing Ops/Sec*	Earth Orbital Operations Memory 32 Bits	Earth Orbital Operations Ops/Sec*	Lunar Orbital Operations Memory 32 Bits	Lunar Orbital Operations Ops/Sec*	Unmanned Planetary (1) Memory 32 Bits	Unmanned Planetary (1) Ops/Sec*	Unmanned Planetary (2) Memory 32 Bits	Unmanned Planetary (2) Ops/Sec*	Reusable Nuc.** Shuttle (RNS) Memory 32 Bits	Reusable Nuc.** Shuttle (RNS) Ops/Sec*	Four Stage Saturn V Memory 32 Bits	Four Stage Saturn V Ops/Sec*
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	30 000	9 600	30 000	5 800	30 000	3 000	15 000	3 000	15 000	9 800	30 000	5 500	30 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	61 800	8 800	61 800	8 800	61 900	8 900	61 800	2 450	61 800	8 800	61 800	8 800	61 800
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	12 [illegible]	1 500	12 500	1 500	12 500	1 500	12 500	3 000	12 500	7 500	12 000	3 500	12 500
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	22 000	2 100	11 800	2 700	17 000	2 700	17 900	2 700	17 800	2 700	17 800	8 200	20 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	1 000	500	1 000	500	2 500	1 000	2 500	1 000	3 000	1 500	7 500	1 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	41 000	4 600	41 000	4 600	41 000	2 300	20 000	2 300	20 000	4 620	41 000	4 600	41 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	16 000	13 400	16 000	13 400	16 000	10 500	16 000	10 500	16 000	13 400	19 070	13 400	19 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0	8 600	50 000	[illegible]	50 000	8 600	50 000	0	0	0	0	11 600	50 002	8 600	50 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	4 000	1 000	4 000	1 000	4 000	1 000	4 000	1 000	4 000	2 000	8 000	1 000	4 000
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	51 800	236 700	51 800	236 700	34 400	148 200	27 450	148 200	54 600	241 720	52 600	239 300
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	7 540	35 505	7 540	35 505	5 160	22 230	4 118	22 230	8 190	30 255	7 890	35 895
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	59 340	272 [illegible]	59 340	272 205	39 560	170 430	31 568	170 430	62 790	271 975	60 490	275 195

[illegible]

Table 4-1. Memory and CPU Requirements

[illegible]	SYNC ORBIT (EXP. VEHICLE)		SYNC ORBIT (1) REUSABLE VEH		SYNC ORBIT (2) REUSABLE VEH	
	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	0
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

[illegible]	LUNAR LANDING		EARTH ORBITAL OPERATIONS		LUNAR ORBITAL OPERATIONS	
	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*
[illegible]	5 600	30 000	5 800	30 000	5 900	30 000
[illegible]	11 000	61 800	8 000	61 950	8 800	61 900
[illegible]	1 400	[illegible]	[illegible]	12 500	1 500	12 900
[illegible]	[illegible]	22 000	2 700	11 800	2 100	17 900
[illegible]	1 200	[illegible]	1 000	500	1 000	500
[illegible]	[illegible]	41 000	4 600	41 000	4 000	41 000
[illegible]	11 600	19 000	11 600	19 000	11 400	19 000
[illegible]	[illegible]	50 000	6 000	50 000	6 100	50 000
[illegible]	1 000	4 000	1 000	4 000	1 000	4 000
[illegible]	51 000	241 500	51 800	236 700	51 800	236 700
[illegible]	[illegible]	35 275	[illegible]	35 505	7 740	35 505
[illegible]	58 470	277 125	58 580	272 480	58 340	272 205

[illegible]	UNMANNED PLANETARY (1)		UNMANNED PLANETARY (2)		REUSAB. & NUC.** SHUTTLE (RNS)		FOUR STAGE SATURN V	
	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*	MEMORY 32 BITS	OPS/SEC*
[illegible]	3 000	15 000	1 000	15 000	5 600	10 000	5 900	30 000
[illegible]	9 000	61 800	2 450	61 800	8 900	61 800	8 900	61 200
[illegible]	1 500	12 500	3 000	12 500	1 500	12 900	3 500	12 900
[illegible]	2 700	11 000	2 700	11 900	2 700	11 900	6 200	20 000
[illegible]	2 500	1 000	2 500	1 000	1 000	1 500	2 500	1 000
[illegible]	7 300	20 000	7 300	20 000	4 000	41 000	4 000	41 000
[illegible]	10 500	16 000	10 500	16 000	13 400	15 000	13 400	19 000
[illegible]	0	0	0	0	6 600	50 000	8 000	60 000
[illegible]	1 000	4 000	1 000	4 000	3 000	8 000	1 000	4 000
[illegible]	34 400	148 200	27 450	148 200	54 000	241 700	52 000	230 300
[illegible]	5 190	22 230	6 118	22 230	8 100	36 755	7 890	35 095
[illegible]	39 560	170 430	11 568	170 430	62 100	277 955	60 490	275 195

[illegible]

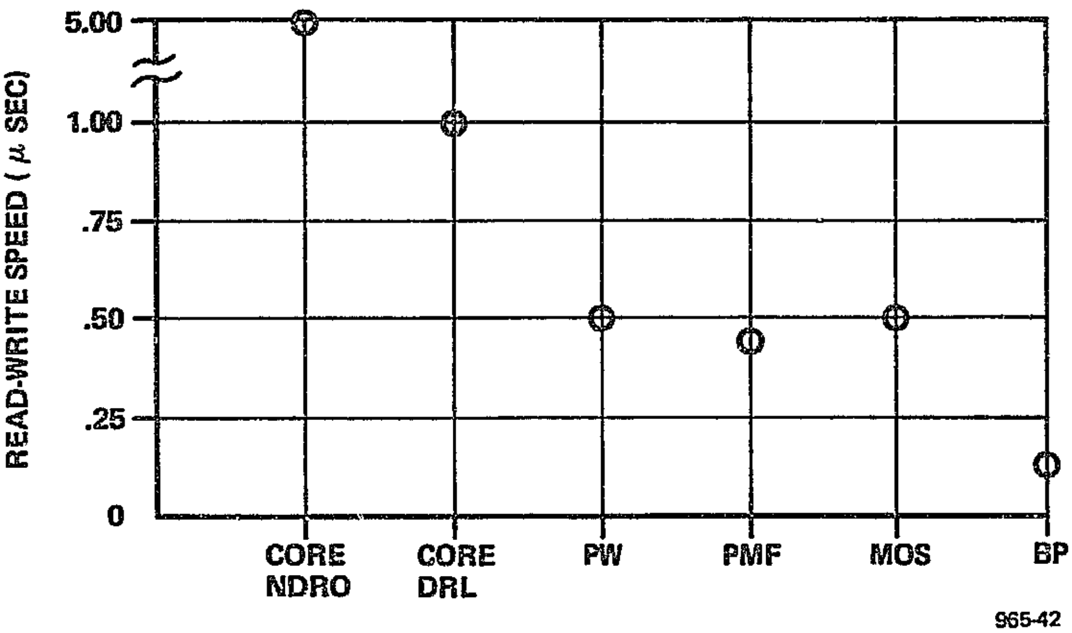

Figure 4-2. Memory Speed Technology

4.1.1.2 Reliability

MTBF's projected for the early 1970's indicate that, while magnetic memories are somewhat more reliable than memories using monolithic technology, implementation techniques are available that can increase the overall reliability of monolithic memories sufficiently to make them candidates for use in space tug. One such technique that lends itself well to monolithic implementation uses several basic operating memories (BOM) (see reference C-2) each containing single bit words (bit-per-BOM). For space tug purposes, each BOM would probably contain 32K x 1 bit words. The total memory unit would then be made up of 32 BOMS for data storage, and 7 BOMS for error correction coding (ECC), or a total of 32K x 39 bit words. The 7 ECC bits will detect and correct one error in each data path and detect a second error in each path. In addition, spare BOM's may be used to replace operating BOM's having a significant number of defective bits. This technique handles single bit transient errors well in that the error is corrected without circuit switching: and once the transient has passed, operation continues as though no error had occurred.

One disadvantage of using monolithics is that while several errors may be detected and corrected, the fact that monolithics have a higher failure rate than magnetic devices indicates a probable increase in maintenance. Another disadvantage with monolithics is its volatility characteristic, i.e., when power is taken off the monolithic memory, it does not retain its state and must therefore be reloaded when power is returned.

4.1.1.3 Power, Weight, and Volume

Figure 4-3 shows projections for memory technology, weight, power, and volume requirements for the early 1970's based on 8K x 36 bits. Thus, Figures 4-2 and 4-3 provide comparisons of cycle time, weight, power, and volume for making random access memory trade studies.

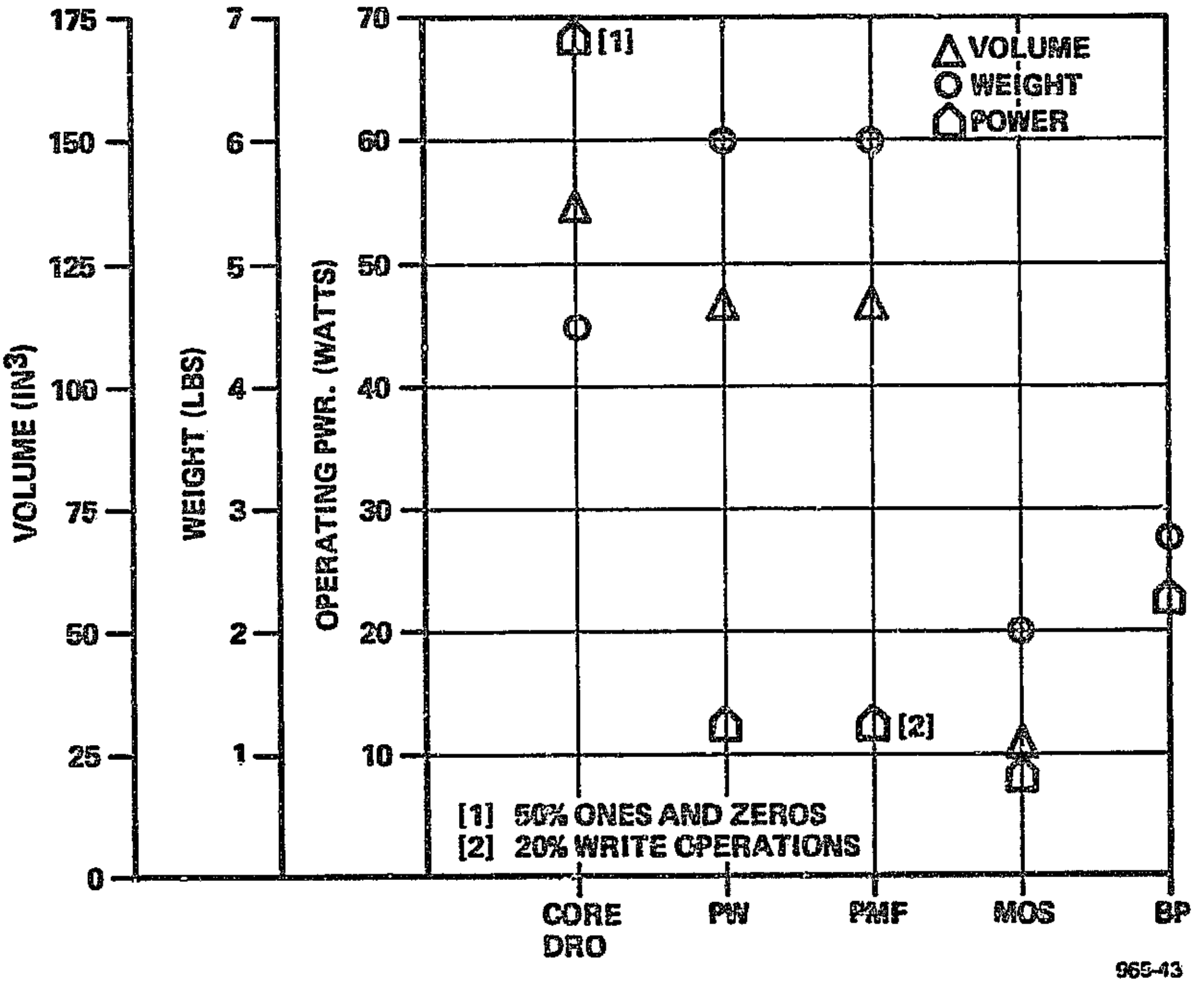

Figure 4-3. Memory Technology Power. Weight and Volume

4.1.1.4 Recommended Approach

The advantages of low power. weight. and volume tend to make monolithic technology a more attractive choice than magnetic devices for implementing space tug random access memory. However, the probable increase in maintenance with monolithic technology and the monolithic volatility characteristic tend to make magnetic devices more attractive.

Significant progress is being made to overcome the volatility problem. and it is probable that for the space tug time frame this problem will be solved. However, it is still not clear at this time which hardware technology or memory organization technique should be used for space tug. While the bit-per-BOM memory organization technique lends itself well to monolithic technology: it is not very compatible with memory addressing circuitry used with magnetic device technology. Thus, it is not a simple matter of choosing the best memory technology and best memory organization technique and then putting the two together. A more detailed study should be made that reflects not only trades within hardware technologies and trades within memory organization techniques: but also trades between combinations of these.

4.1.2 Bulk Storage

The required quantity of bulk storage or non-random access memory for space tug will range from none for the expendable synchronous earth orbit mission to several hundred thousand 32-bit locations for the more complex manned missions. Bulk storage will be used for some missions to (1) redundantly store all programs that are used in the random access memory, (2) store extensive checkout and diagnostic programs, and (3) store programs for refueling support. There are probably other requirements for bulk storage, such as recording some mission data, that are not presently defined. However, the massive amount of storage contained in a basic bulk storage unit should be capable of accommodating these added requirements.

The currently available non-random access memory technologies are as follows:

- Drum memories
- Disc memories
- Magnetic tape

The disc system has a basic technology problem of obtaining the necessary disc flatness (about $+20 \times 10^{-6}$ inches) and then maintaining it over temperature ranges and during system vibration.

Magnetic tape is very attractive for large volume data storage and for data readout if the computing system can be efficiently oriented towards a sequential data source. This is the case when using a tape for memory loading. However, for reading non-sequential information, the magnetic tape is extremely slow. From a cost per bit, weight, power, and volume standpoint, magnetic tape is more attractive than the rotating drum system. The magnetic drum has an advantage over tapes for non-sequential information reading with access times several orders of magnitude better than the magnetic tape. Table 4-2 shows the more important characteristics of the three non-random access storage devices mentioned above.

Table 4-2. Characteristics of Non-Random Access Storage Devices

CHARACTERISTIC	DRUM	MAGNETIC TAPE	DISC.
STORAGE SIZE (BITS)	16.0×10^6	20.6×10^6	8×10^6
WEIGHT (LBS.)	50.0	13.5	120.0
VOLUME (IN.^{3})	1970	350	3880
POWER (WATTS)	240	50	130
COST/BIT (CENTS)	LESS THAN .375	LESS THAN 0.01	LESS THAN .10
RELIABILITY (MTBF, HRS)	10,000	13,000	NOT AVAILABLE
AVERAGE ACCESS TIME (MSEC)	~8.0	~500.0	~75.0

The only bulk storage requirement showing a need for fast access time is that of display skeletons. This requirement can be handled by a separate monolithic or magnetic type memory (32K of 32-bit words); and if the number and types of skeletons are fixed for a given vehicle, the memory could be a "read only" type, thus eliminating the necessity for memory reloading, resulting in lower power, weight, and volume for this memory. With the fast access requirement accommodated by a separate storage unit, the choice for bulk storage is the magnetic tape unit.

4.1.3 Central Processing Unit (CPU)

The number of CPU operations/second necessary to perform the required functions for each space tug mission is shown in Table 4-1. As shown, requirements for the various space tug missions result in a wide range of speed requirements, i.e., the manned missions require extensive checkout and diagnostic routines and display support programs to be executed in real-time by the CPU; whereas, an unmanned expendable mission requires no display support and a reduced real-time checkout operation. Thus, operations/second requirements are reduced considerably for the latter mission, and a less powerful CPU will meet these requirements. In selecting CPU's for space tug, the options considered were as follows:

- For all missions, use one CPU of medium speed capability (300K–400K ops/sec.) that will accommodate the most severe mission case. This option tends to minimize the required changes in hardware and software for the whole range of space tug missions; and it tends to reduce synchronization problems inherent with use of several dedicated CPU's. However, for missions using expendable space tugs, this option has the disadvantage of having to use a computer that weighs more, costs more, and consumes more power than a less powerful CPU that could also handle mission requirements.

- For all missions, use several less powerful CPU's. Thus, for missions having less requirements and using expendable space tugs, only the number of CPU's needed to meet requirements would be used. This is feasible with the modular astrionics approach. However, for missions with greater requirements, it complicates software changes, verification and maintenance; and it also makes total system synchronization, checkout, error correction and maintenance more complex.

- Design a family of computers with compatible software and operation which will handle the range of missions while providing a near-optimum CPU/mission match. Thus, the CPU best suited for a particular mission would be used. The disadvantage of this approach is the cost of developing such a family of CPU's, the problem of ensuring compatibility between family members, and the problem of adapting each to the modular structure.

At this time, the first option (one CPU of medium speed capability) appears to be the best choice for space tug. This final choice will also depend on the best choice for the other space elements, such as RNS and space shuttle. While CPU capability may be wasted for some tug missions using this option, the overall reduction in complexity makes it attractive; and it is expected as more detailed trade studies are made that this advantage will be seen to outweigh the advantages of other options.

The numbers shown in Table 4-1 are intended to reflect the most stringent requirements for each space tug mission; thus, the CPU speed specification was based on this table. However, it is possible that the refueling support requirement may be of such complexity that the speed requirements on the CPU will be more severe than the real time requirements for any mission. This refueling requirement has not been defined well enough at this time to make an evaluation, but should be addressed as soon as information is available. Until such time, it will be assumed that Table 4-1 reflects worst case CPU requirements.

Based on requirements for the several missions, the space tug CPU should have the following design goals: be in the 300K operations/second or greater range; weigh 30 pounds or less; have power requirements of 30 watts or less; have a 128 instruction repertoire that is compatible with a ground-based computer, thus easing software development and verification support. In addition, the CPU should have floating-point operation capability and be a parallel machine. These goals are somewhat more stringent than the goals required by an earlier report (see reference C-3). This is due to requirements being better defined at this time. CPU's using extended state-of-the-art technology are being developed that will meet most of these goals. New technology currently under development should easily meet all these goals in the space tug time frame.

4.1.4 Bus Control and I/O Unit (BCIOU)

The Bus Control and I/O Unit (BCIOU) will serve to control all information flow between the CPU and the data bus and between the random access memory and the data bus. It will send and receive address and data information to and from the data bus in serial digital form. It will receive data and address information from the CPU in parallel digital form via a Program Controlled Output (PCO) channel or Externally Controlled Output (ECO) channel where external control is performed by the BCIOU. Data is output from the random access memory whenever a data request and memory address is input from the BCIOU via the ECO channel. These alternate output paths provide the option to completely control the flow of data request and data outputs with program execution (PCO channel) or to allow the BCIOU to control the output via the ECO channel once it has been initiated by the CPU via the PCO channel. A combination of these two approaches seems to offer the greatest flexibility for output control.

The CPU will store in predetermined memory locations any desired sequence of addresses and data that are to be sent to the data bus system; it then initializes execution of the sequence via the PCO channel. Using the ECO channel then, the BCIOU will sequentially fetch the stored addresses and data from memory and transfer them to the data bus serially. Sequencing will continue until the last address in the sequence is processed or until a termination command is received from the CPU via the PCO channel. The last word in the sequence will be a special I/O control word which commands the BCIOU to terminate the sequence processing. Control words will also be used to clear the BCIOU, initialize I/O block transfers, and modify the ECO channels' storage address register to allow branching. Once initialized, the BCIOU will perform all of the control functions necessary to complete transfer with the data bus systems.

Input data will be transferred to the memory via an Externally Controlled Input (ECI) channel. This is done by the BCIOU requesting a memory cycle and specifying the desired storage address.

4.1.5 System-to-System Communication

4.1.5.1 Introduction

The trend (see reference C-4) in the design of large astrionic systems is to incorporate a data bus to provide exchange of information between the various astrionic subsystems rather than use a centralized approach where subsystems link directly with a central computer or with each other. The data bus concept, its advantages, and the problems attendant to its implementation are summarized in the following paragraphs.

The salient feature of the data bus is a standard digital interface that can be specified for components performing varied functions in large and complex astrionic systems. In addition, a data bus offers to an astrionic system the advantages of flexibility, low weight, high reliability, and ease of maintenance.

Flexibility results because subsystems or components can be added, deleted, or upgraded with little or no effect on the system interface. Components access the data bus and obtain the output information they require to perform their designated functions.

The weight advantage (see reference C-4) of the bus concept over the conventional centralized system is a result of multiplexing. In the conventional centralized system, numerous wires carry signals to and from the various system components in a variety of forms and on many cables. In a bus system, all data to and from the components are carried on one or a small number of cables.

Reliability is an advantage that can be realized because interface circuitry, wiring, and connectors are reduced and more importantly, through a more efficient system functional design. Checkout is enhanced because the data bus serves as a single test point that allows access to hardware through a centralized integrated testing point. It enables a central digital computer or special test device to test and monitor performance before and during powered flight.

Maintenance is simplified by eliminating signal distributors and multiple connectors, by standardizing a communication technique, and by providing ready access to components that need to be added or deleted without impacting the entire system.

Many of the advantages of the data bus system are made possible through the techniques of multiplexing. A technique that addresses serial digital data which is to be transferred through time division multiplexing on two twisted shielded pairs is described here. The major influencing factor in determining the architecture for a time division multiplexed bus system is the method of controlling access to the bus. Uncontrolled random access would permit multiple units of data to be transmitted simultaneously with the high probability of scrambling of data.

There are at least three general techniques (see reference C-4) to controlling bus access. They are defined and summarized as follows:

(1) Central Control Approach — Under this design concept, a central control unit or data adapter is programmed to sample all input/output devices at a predefined rate that is great enough to satisfy system dynamics and operation. Units on line

respond only upon request; hence, bus control is maintained by the central controller. This concept should result in simple input/output device interface circuits, but would also result in a relatively complex controller with several stored sampling formats.

(2) Interrupt Approach – Under this design concept, input/output devices flag the central controller when it has information ready or when it requires attention. The processor then addresses the input/output device to allow information transfer. Bus control is also maintained by the central control unit. Most centralized digital computer controlled systems (A7, ATMDC, LVDA, etc.) use this approach which can also be implemented for bus control. Advantages are lower average transport lag time than the central control approach at the cost of requiring an independent interrupt function. Independent data transfers between input/output devices are not possible.

(3) Polling Approach – Under this design concept, a polling signal circulates through each unit on the bus. An input/output device acquires access to the data bus by capturing the polling signal. A somewhat random access is thereby possible without requiring attention from a central control unit. The advantage of the concept lies in automatic time multiplexing of data indpendent of the central controller and input/output device response time.

4.1.5.2 Functional Operation

The desirable functional characteristics of any data bus implemented for an avionic or astrionic system should include the capability for the following:

(1) Any input/output device to issue or receive commands and data from any other device in the system independent of a central flow point.

(2) Any input/output device to randomly access the transmission media.

(3) A master controlling element on the bus to govern and, if necessary, restrict the data transmission.

Any system architecture which achieves these characteristics should also accomplish the following.

(1) Insure the reliability of the information transmissions.

(2) Standardize and minimize the bus control logic within each input/output device.

(3) Reduce the cabling or the transmission media to a minimum.

(4) Permit the addition or deletion of input/output devices to or from the system without adversely affecting interfaces or control logic.

4.1.5.3 Data Bus Selection

At this phase of the study, the data bus technique that will probably be chosen for space tug is the central control approach, which uses two pairs of twisted shielded wires terminated at the end of the bus. One of the pairs, called the data line, transfers digital data serially in a two-way fashion: the other pair, called the address line, provides multiplexing control by designating which device will send data and which device will receive it. The bus originates in and is controlled by a bus control unit which is interconnected with a digital computer. The bus control unit provides the addresses that allow one device to communicate with any other device and directs the flow of information between the computer main storage and each input/output device attached to the bus. The control unit relieves the central processing unit of the tasks of communicating directly with input/output devices and permits data processing to proceed concurrently with input/output operations. The single data path of the bus is time-shared by the input/output devices which operate, as commanded, by addresses from the bus control unit. The physical and electrical connection between each input/output device and the bus is called the standard interface unit. All devices, regardless of their differences, can be added to the system without hardware changes or addition of new instructions to the central processor, provided they satisfy the requirements on their side of the interface. Figure 4-4 is a simplified functional diagram of the two pair bus system.

4.1.6 Data Bus Standard Interface Unit

The standard interface unit adapts the input/output unit of each device to the data bus. The functional circuitry is shown on the diagram of Figure 4-5.

The standard interface unit is identical for all monitored units. It consists of the circuitry necessary to decode the digital address of the measurement or command, receive the command or data from the data bus or deliver data to the data bus. The unit receives serially formatted digital data from the data bus and makes it available to the commanded location in serial, parallel, or analog form as required by the device. In like manner, the unit receives serial, parallel, or analog data from the addressed location and transmits the data in serially formatted digital form onto the data bus.

By employing the standard interface unit in this form, each device can use known standard multiplexers and signal conditioners that are compatible with that type of circuitry. At the same time, the standard interface unit, utilizing latching relays or sample and hold circuits, performs all the functions of the switch selector, thus eliminating the requirement for that unit. The use of the standard interface unit and formats allows the design of each device to be indpendent of the design of any other device.

In operation, the standard interface unit receives the digitally coded address from the bus address line, determines that the signal is intended for its device, and turns on the appropriate multiplexer and sequences into the proper channel. The addressed standard interface unit will then either receive data from or transmit data to the data bus after the data has been properly converted by the A/D or D/A converters. Two parallel storage registers will be provided to temporarily store parallel data while the shift register on the A/D ladder is engaged with time dependent tasks. The data will be strobed into and out of the parallel storage registers to reduce the cycle time and bus dead time. It should be possible to obtain nearly 100% duty cycle usage of the data bus.

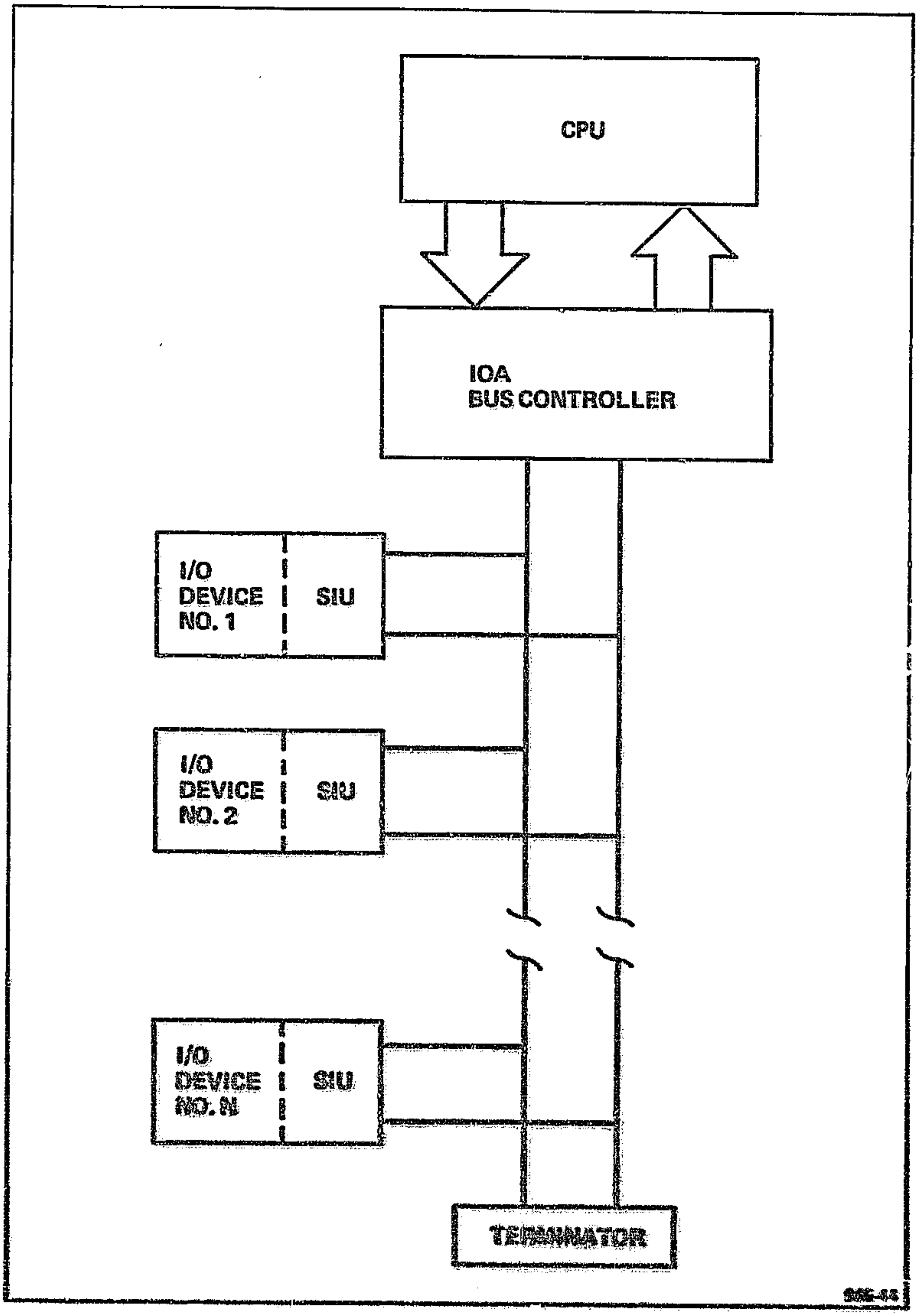

Figure 4-4. Simplified Functional Diagram of a Two-Wire Data Bus System

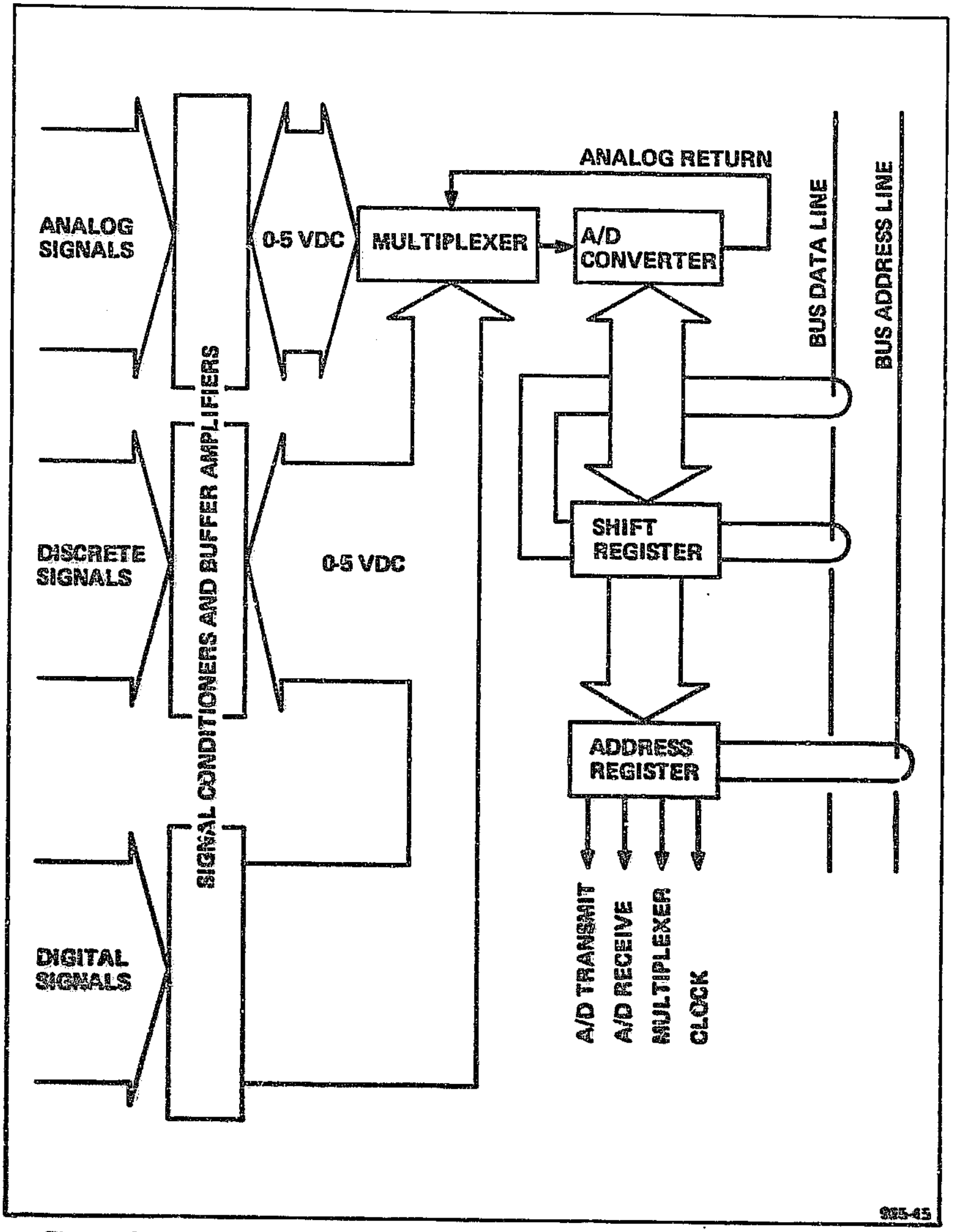

Figure 4-5. Standard Interface Unit and Input/Output Device

4.1.7 Input/Output Unit

The input/output unit adapts and multiplexes the various measurement data and commands between the device and the standard interface unit. The functional circuitry is also shown on the diagram of Figure 4-5.

The input/output unit is unique to the device being monitored and consists of the circuitry necessary to gather or disseminate all measurements, sequencing commands, discretes, or digital data. The unit receives voltage signals from various parts of the device being monitored and converts them to a normalized form by signal conditioning. These signals are then multiplexed on command to serial, parallel, or analog waveforms to be further controlled by the standard interface unit. In like manner, the unit will receive serial, parallel, or analog data from the standard interface unit and demultiplex them — on command — for application within the device.

By employing the input/output unit in this form, the device design can utilize multiplexing and signal conditioning compatible to that type of design with very few stringent requirements on the output or input data.

In operation, the input/output unit is energized at the time the standard interface unit determines that a request or command is needed in the device. The standard interface unit also programs the multiplexer to the proper channel. The input/output unit then will deliver a conditioned signal or signals to or accept a signal or signals for signal conditioning from the standard interface unit.

The number of system measurements has been assumed to be approximately equal to an Apollo Saturn S-IVB, IU, and CSM measurement system.

Twenty percent of these measurements are assumed to emanate from packaged devices which contain their own input/output unit. The remainder of the measurements will be separately multiplexed in a manner similar to an input/output unit and connected to the data bus by a standard interface unit. There are, therefore, 9 multiplexers for the astrionic module, 3 multiplexers for the propulsion module, and 6 multiplexers for the crew module.

The power requirements for the measurements portion is therefore assumed to be distributed as shown in Tables 4-3 and 4-4.

Table 4-3. Space Tug Measurement Requirements

MEASUREMENTS	ASTRIONIC MODULE	PROPULSION MODULE	CREW MODULE
TOTAL MEASUREMENTS	1 500	440	1 000
AC AND DC (60%)	900	264	600
DISCRETE AND DIGITAL (40%)	600	176	400
SIGNAL CONDITIONERS*	450	132	300

*FIFTY PERCENT OF THE AC AND DC SIGNALS ARE ASSUMED TO REQUIRE SIGNAL CONDITIONING

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MEASUREMENTS	ASTRIONIC MODULE	PROPULSION MODULE	CREW MODULE
TOTAL MEASUREMENTS	1,500	440	1,000
AC AND DC (60%)	900	264	600
DISCRETE AND DIGITAL (40%)	600	176	400
SIGNAL CONDITIONERS*	450	132	300

*FIFTY PERCENT OF THE AC AND DC SIGNALS ARE ASSUMED TO REQUIRE SIGNAL CONDITIONING

Table 4-4. Measurement Power Requirements

MEASUREMENTS	ASTRIONIC MODULE	PROPULSION MODULE	CREW MODULE
AC AND DC (2 MA EACH)	1,800 MA	528 MA	1,200 MA
DISCRETE (1 MA EACH)	600 MA	176 MA	400 MA
SIGNAL COND. (10 MA EACH)	4,500 MA	1,320 MA	3,000 MA
TOTAL	6,900 MA	2,024 MA	4,600 MA
NORMAL CURRENT (30%)	2,070 MA	607.2 MA	1,380 MA
PEAK CURRENT (60%)	4,140 MA	1,214.4 MA	2,760 MA
NORMAL POWER	56.2 W	17.0 W	44.6 W
PEAK POWER	112.4 W	34.0 W	89.2 W

4.1.8 Configuration Assignment Unit (CAU)

The Configuration Assignment Unit (CAU) provides the means for recovery after an error is detected in the main storage, CPU or the bus control and I/O unit (see reference C-7). The initial step of recovery is provided by the use of interface error detectors which test a device output dynamically and which are self-testing themselves. These are used to provide error alarm signals to the CAU and it therefore becomes the prime responsibility of this subsystem to react to the error condition. Basically, then, the CAU can be subdivided into two subfunctions, bootstrap recovery and diagnosis. Through the bootstrap recovery subfunction, the CAU serves four purposes. It first provides a nontrivial and complex interface between the failure detection hardware and the diagnosis programs. Second, it provides a means for rapid retry capability for recovery from transients. Third, it provides a system capacity for ascertaining the existence of catastrophic failures. And, finally, it initiates system reconfiguration when necessary.

4.2 DATA MANAGEMENT ORGANIZATION, REDUNDANCY AND FAILURE DETECTION

The organization, redundancy and failure detection was approached with goals of (1) minimizing total weight, power consumption, and volume, (2) minimizing impact of adding or deleting components or subsystems to accommodate the various missions, (3) minimizing impact of software modifications by minimizing the required verification effort while maintaining confidence in sufficiency of effort, (4) minimizing checkout and component switching effort, and (5) maximizing system reliability, crew safety, and probability of fulfilling all objectives.

Figure 4-1 shows the data management organization for the unmanned synchronous orbit mission. As mentioned earlier, the random access memory (RAM) organization and redundancy will depend on the hardware technology used. If monolithic technology is

employed, the RAM will probably consist of BOM's of 32K words with 1-bit per word. Thirty-nine of these BOM's will be combined to provide 32K of 39-bit words (32-bit data and 7-bit error code). One standby BOM will be included to provide capability to replace a failed BOM or one with a significant number of defects. However, if magnetic devices are employed, the bit per BOM organization will not be used. Further trade studies will determine this.

Memory control and interface will provide capability for each 32K memory to be controlled (read or store) by the CPU's (primary or backup), bus control and I/O units (primary or backup), and the configuration assignment unit.

For the synchronous orbit mission, two CPU's will be used; one as the operating primary and the other as the standby backup. This provides a fail safe capability. Manned missions will require two backup CPU's and two backup bus control and I/O units to provide a fail operate/fail safe capability. The CPU's in conjunction with the configuration assignment unit (CAU) provide hardware to detect failures in the CPU, bus control and I/O units, and memory; determine if they resulted from transient or hard failures; and switch to the backup CPU, the backup bus control or backup memory module, as required.

The communication between the CPU's and bus control and I/O units will consist of: (1) one program controlled output (PCO) channel, (2) one externally controlled output (ECO) channel, (3) one externally controlled input (ECI) channel, which actually links the memory with the bus control and I/O units, and (4) several interrupts.

Each CPU will have a small dedicated non-volatile memory to provide bootstrap startup and loading after the data management subsystem has been powered down.

For the synchronous orbit mission, the data bus and each operating standard interface unit (SIU) will be provided with one standby backup to meet the fail/safe criterion. Manned missions will require that the data bus and the SIUs handling critical functions be provided with two backups to meet the fail operate/fail safe criterion.

All data and addresses transmitted on the data bus will be encoded for error detection purposes.

SIU failure detection will probably be accomplished through self detection hardware techniques. However, trade studies should be made to evaluate both hardware and software techniques for this application.

Sensors and other devices being monitored or controlled will probably be checked for failures through application of software techniques (limit checking, reasonableness tests, and trend analysis). However, failure detection may be accomplished by adding test circuitry in the SIU's or possibly a combination of hardware and software may be desirable. A trade study should be made to result in the best choice.

As mentioned earlier and shown in Figure 4-1, storage for display formats or skeletons will probably be provided by a 32K monolithic or magnetic memory for manned missions.

The lunar mission will include an auxiliary monitor computer to support monitoring and displaying of critical parameters to the crew during the 28 or 42 day period on the lunar surface. This allows the DMS to be powered down during this period and also allows DMS reliability to be maintained at a high level for the entire mission.

As shown in Figure 4-1, sequencing will be performed via the same SIU-I/O units that provide data monitoring and measurements in each vehicle module. Two concepts were considered for performing the sequencing function: (1) use separate switch selectors as in Saturn program and (2) let SIU-I/O address logic handle sequencing commands. In the former case, a separate switch selector type device would be employed in each module (propulsion, crew, secondary propulsion, and astrionic module). In the latter case the sequencing commands would be handled by the SIU-I/O address logic just as other information (data and discretes) are handled. However, just as with the switch selector, some means of read back verification by the computer will be required. Thus, both concepts are much the same; the difference being that in the first concept, the sequencers are separate units and in the latter concept, the sequencers are integrated with other information conveying hardware which multiplexes all incoming and outgoing information. The latter concept was chosen because of the reduction in hardware.

Table 4-5 shows a summary of the equipment required for each mission, and Table 4-6 summarizes power, weight, and volume requirements for each data management subsystem component.

4.3 SOFTWARE ORGANIZATION AND REQUIREMENTS

Two types of software organization were considered for space tug application:

(1) Conglomerate Organization in which a single complete software package is written for each mission or phase between missions. Subroutines performing various functions are interlaced to minimize the initial programming effort; thus, subroutines are highly dependent on each other.

(2) Modular Organization in which function modules are written to perform single functions such as navigation, guidance or control for a particular mission or phase between missions. Function subroutines are written in modular form so that each function module is isolated as much as possible from all others. Thus, an executive control program will load from bulk storage into main memory only those function modules required to perform a particular mission or phase.

The modular rather than the conglomerate software approach will probably be used for implementing the space tug software package. The main reason for this is the fact that unlike Saturn, which has a lifetime of only one mission and thus requires only one software configuration, space tug will have a lifetime consisting not only of many missions but also of many phases between missions, all of which will require different flight software configurations. Thus, since only one software configuration is required for Saturn, either the conglomerate or the modular approach can be used with little difference in software size and verification effort. However, if the conglomerate approach is used with space tug, a different software package will be required for each mission and for each phase between

Table 4-5. DMS Equipment List

COMPONENT	MISSION																													
				SYNCHRONOUS ORBIT																PLANETARY										
	SYNCHRONOUS ORBIT (EXPENDABLE STAGE)			(REUSABLE FIRST STAGE)			(REUSABLE SECOND STAGE)			LUNAR LANDING			EARTH ORBITAL OPERATIONS			LUNAR ORBITAL OPERATIONS			(REUSABLE FIRST STAGE)			(EXPENDABLE SECOND STAGE)			RNS			FOUR STAGE SATURN V		
	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O	A	S	O
CPU	1			1	1		1	1		1	1	1	1	1	1	1	1	1	1	1		1	1		1	1		1	1	
BUS CONTROL UNIT	1			1	1		1	1		1	1	1	1	1	1	1	1	1	1	1		1	1		1	1		1	1	
MAIN MEMORY (40 BOM-32K)	1			2			2			2			2			2			2			2			2			2		
MAGNETIC TAPE UNIT				1			1			1			1			1			1			1			1			1		
SIU	19			27	5	25	25	5	23	36	6	33	35	5	33	35	5	33		5			5			5			5	
DATA BUS	1			1	2		1	2		1	2		1	2		1	2		1	2		1	2		1	2		1	2	
PRIMARY MONITORING UNIT	12			12		12	12		12	18		18	18		18	18		18	12		12	12		12	18		18	18		18
DISPLAY MEMORY (40 BOM-32K)										1			1			1									1			1		
CONFIGURATION ASSIGN. UNIT				1	1		1	1		1	2		1	2		1	2		1	1		1	1		1			1		
AUX. MONITOR COMPUTER										1																				
AUX. MONITORING UNITS*										3																				

* USED ONLY WHILE SPACE TUG ON LUNAR SURFACE.

A = ACTIVE
S = STANDBY
O = OFF

Table 4-6. Component Characteristics

COMPONENT	COMPONENT CHARACTERISTICS							
	DIMENSIONS (IN)			WT. (LBS)	POWER (W)		COOLING LOAD	TEMP. LIMITS
	W	H	D	EACH	A*	S*	BTU/HR/UNIT	°F
CPU	13.4	9.4	7.0	60.0	80.0	80.0	273.0	-40 TO 131
BUS CONTROL UNIT	12.0	8.0	3.6	35.0	50.0	50.0	170.7	-40 TO 131
MAIN MEMORY (40 BOM-32K)	10.0	6.0	5.0	20.0	50.0	5.0	170.7	-40 TO 131
MAGNETIC TAPE UNIT	10.0	7.0	5.0	10.0	15.0	0.0	170.7	32 TO 130
SIU	4.3	3.0	2.8	0.4	1.0	1.0	3.4	-67 TO 257
DATA BUS	173.0 IN.^{3}			7.5	—	—	—	-40 TO 131
MONITOR. UNITS	4.0	2.0	2.0	0.5	2.0	2.0	6.8	-40 TO 131
CONFIG. ASSIGN. UNITS	8.0	6.0	3.0	12.0	15.0	15.0	51.2	-40 TO 131
DISPLAY MEMORY (40 BOM-32K)	5.0	6.0	5.0	10.0	50.0	5.0	170.7	-40 TO 131
AUX. MONITOR. COMPUTER	7.0	5.0	3.5	15.0	20.0	20.0	68.3	-40 TO 131
AUX. MONITOR. UNITS	4.0	2.0	2.0	0.5	2.0	2.0	6.8	-40 TO 131

*A = ACTIVE
S = STANDBY

missions. Whereas, if the modular approach is used, the executive control program will pull from bulk storage and load in main memory only those function modules that are required for the particular phase or mission. It becomes obvious that not only will much more software be required for the conglomerate approach but that the verification effort tends to become prohibitive. Figure 4-6 illustrates the difference between the modular and conglomerate software structures and shows the advantages and disadvantages in using the modular approach.

The modular executive control function which is presented in more detail in Figure 4-7 includes interrupt processing, function execution scheduling and control, mission and phase initialization, and memory loading. Since functions within a mission phase must be executed at different rates with different timing precision, function modules, according to their particular timing requirements, are assigned via system macros to operate under a specific sub-program of the control program. The program structure will consist of the following sub-programs.

- Mission Executive: Governs phase-to-phase transition throughout the entire mission.
- Phase Initializer: Initializes master queue/control tables to proper status at the start of each defined mission phase.

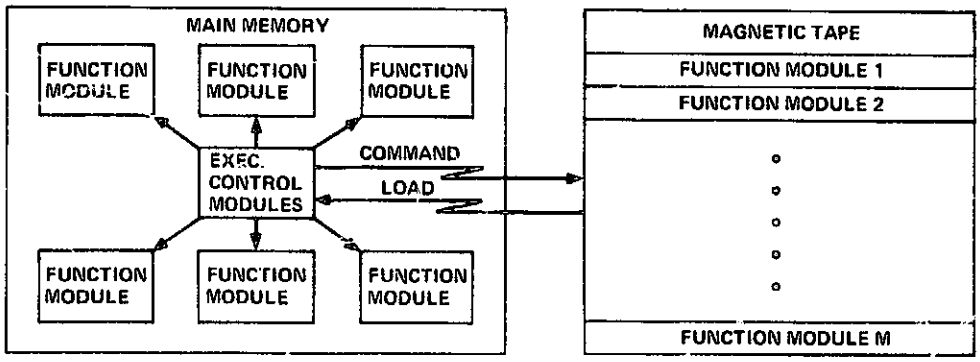
- WHAT IS MODULAR SOFTWARE?

MODULAR SOFTWARE CONSISTS OF TWO BASIC PARTS:

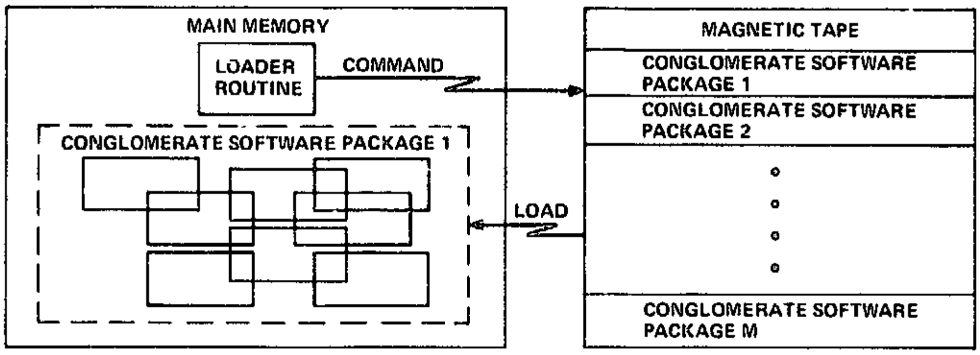
1. FUNCTION MODULES EACH OF WHICH IS RESPONSIBLE FOR PERFORMING A MISSION FUNCTION SUCH AS NAVIGATION, GUIDANCE OR CONTROL FOR A PARTICULAR PHASE OF A MISSION; I.E., THE FUNCTION MODULE USED IS MISSION DEPENDENT.

2. EXECUTIVE CONTROL MODULES WHICH HAVE A PRIMARY PURPOSE OF CONTROLLING THE SEQUENCE AND ORDER OF EXECUTION OF ALL FUNCTION MODULES FOR A PARTICULAR MISSION. THE CONTROL MODULES ARE MISSION INDEPENDENT.

- MODULAR SOFTWARE STRUCTURE



- CONGLOMERATE SOFTWARE STRUCTURE



- ADVANTAGES OF MODULAR SOFTWARE

1. ISOLATES EACH FUNCTION SO THAT THE EFFECTS OF CHANGES IN A GIVEN FUNCTION MODULE WILL BE CONFINED TO THAT MODULE AS MUCH AS POSSIBLE, I.E., OTHER FUNCTIONS SHOULD NOT HAVE TO BE MODIFIED TO ACCOMMODATE THESE CHANGES. THUS, ONLY THE CHANGED MODULE WILL REQUIRE PROGRAM VERIFICATION.

2. NEW MODULES CAN BE ADDED TO THE FUNCTION MODULE POOL WITH MINIMUM IMPACT.

3. MAXIMIZES SOFTWARE RELIABILITY BY CONFINING THE IMPACT OF PROGRAM CHANGES TO THE FUNCTION MODULE BEING MODIFIED.

4. MASS STORAGE IS MINIMIZED SINCE THE ALTERNATIVE TO THE MODULAR APPROACH IS TO HAVE SEPARATE COMPLETE MISSION SOFTWARE PACKAGES FOR EACH MISSION OR PHASE OF OPERATION. IN THIS CASE, SOME FUNCTION ROUTINES WOULD BE CONTAINED IN EACH OF THESE PACKAGES, THUS REQUIRING REDUNDANT STORAGE.

- DISADVANTAGES

1. SLIGHTLY MORE MAIN MEMORY WILL BE REQUIRED DUE TO THE DIFFERENCE IN MODULAR SOFTWARE STRUCTURE.

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Figure 4-6. Space Tug Software

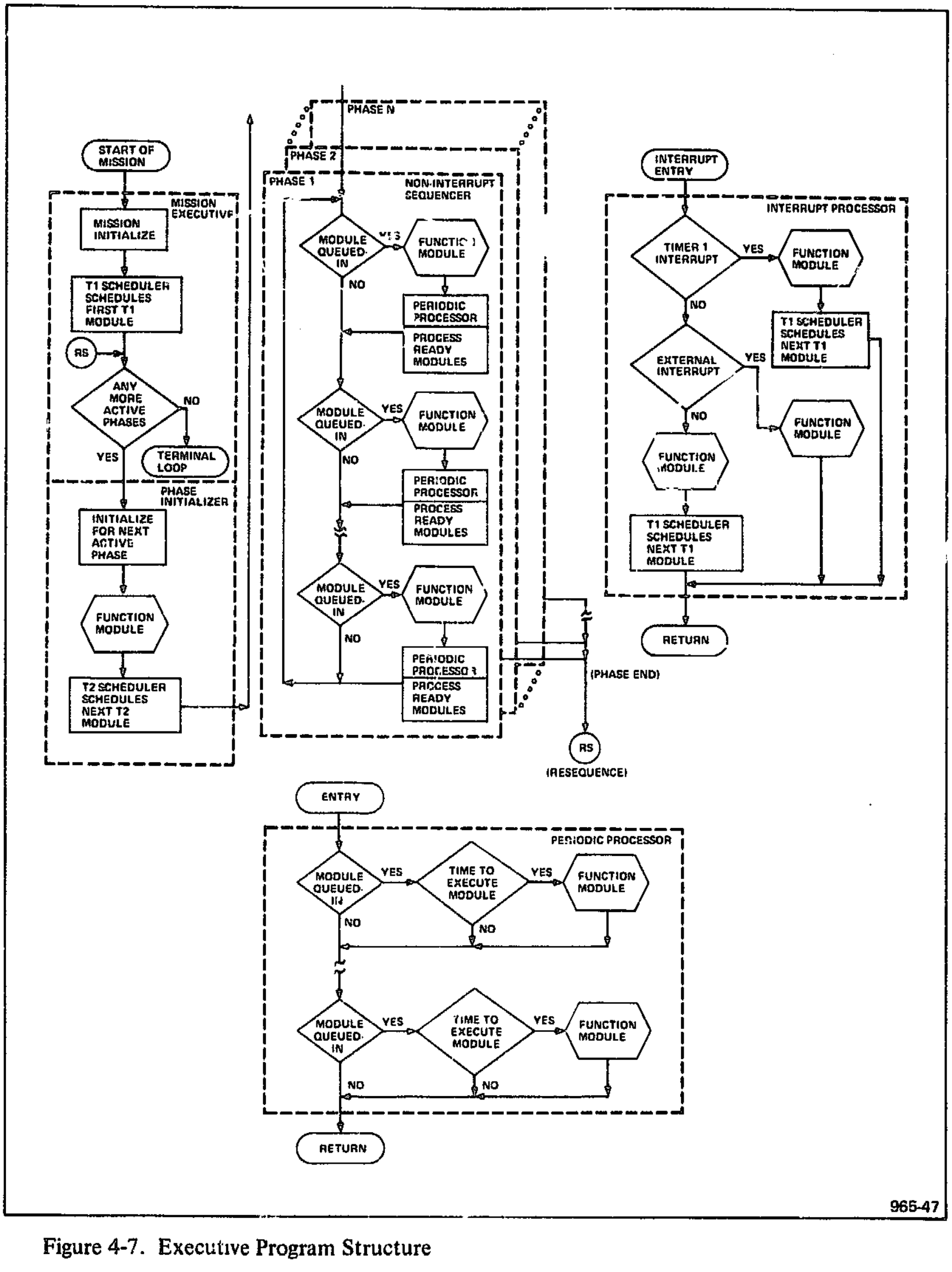

Figure 4-7. Executive Program Structure

- Non-Interrupt Sequencer: Controls the execution of those modules which make up the basic iterative computation of any phase, i.e., those functions of a non-priority nature which are to be operated as a part of a repetitive, ordered sequence.

- Periodic Processor: Controls the execution of those modules of a non-priority nature but which are to be executed a given number of times within a specified, but not precisely rigid, time frame.

- Interrupt Processor: Services both hardware and program controlled interrupts and routes control to the appropriate program module on a priority basis.

- Timed Schedulers: Schedules the use of the timed interrupts to control the execution of priority functions modules which require operation at an exact time or at a precisely cyclic rate.

A study should be made to identify in greater detail the required software function modules for all space tug missions and phases and show how these can be called into main memory as missions and phases change.

4.4 FUNCTIONAL REQUIREMENTS

The details of the data management functions, including flow charts, are presented in other appendices. The purpose of this subsection is to enumerate the data management subsystem functions, show CPU (operations/sec.) and memory (number of locations) requirements per function for each mission. Table 4-1 shows the functions as presently defined along with their respective CPU and memory load requirements. The range of memory requirements for the ten missions shown in Table 4-1 is 31,568 32-bit memory locations for the second unmanned planetary mission to 62,790 locations for the RNS mission. The unmanned planetary mission memory requirement is low primarily because of the lower navigation requirements. It should be noted here that RNS memory and CPU speed requirements do not include nuclear engine control. This is a very complex problem and sufficient information is not available at this time to address these requirements. Based on the information that is available, it appears the problem of nuclear engine control will at least double those memory and speed requirements for RNS presented in Table 4-1.

The range of CPU speed requirements is 153K ops/sec. for the expendable synchronous orbit mission to 278K ops/sec. for RNS. This is somewhat lower than the results of the earlier space tug report (see reference C-6). The difference is due mainly to refinements in navigation and strapdown IMU requirements and a reduction from 25% to 15% for contingencies. The contingencies requirement was reduced because of increased confidence in the refined requirements.

4.5 PROJECTION OF TECHNOLOGY DEVELOPMENT

Table 4-7 (see reference C-7) shows the expected computer technology characteristics resulting from extended state-of-the-art technology in the 1972-1975 time frame. In addition, the expected computer characteristics of new technology development are shown for the post-1974 time frame.

Table 4-7. Projected Technology Development

COMPUTER	CENTRAL PROCESSING UNIT				MEMORY					INPUT/ OUTPUT		PHYSICAL			
	NUMBER OF INSTRUCTIONS	ADD TIME (MICROSECONDS)	MULTIPLY TIME (MICROSECONDS)	THOUSANDS OF OPERATIONS, PER SECOND (1)	WORD SIZE DATA BITS/ PARITY OR OTHER	CAPACITY K = 1024 BASIC	CAPACITY K = 1024 MAX. ADDRESS	ACCESS (MICROSECONDS)	CYCLE (MICROSECONDS)	NUMBER OF CHANNELS	NUMBER OF INTERRUPTS	WEIGHT (POUNDS)	SIZE (CUBIC FEET)	POWER (WATTS)	MTBF (K = 1000 HOURS)
EXTENDED TECHNOLOGY PRODUCTION 1972-1975	75	1.5	4.5	475	32/4	8K	64K	0.5	1.0	2	16	30	0.4	60	15K
NEW TECHNOLOGY PRODUCTION POST 1974	100 VARIA-BLE (2)	0.5	3.0	1000	32/4	16K	128K	0.1	0.3	3	32	20	0.25	30	50K

(1) 80% ADDS, 20% MULTIPLIES
(2) THE INSTRUCTION REPERTOIRE WILL BE VARIABLE THROUGH AN ALTERABLE ROS CONTROL

The weight, power and size given include the basic memories listed, i.e., 8K words for extended technology and 16K words for new technology.

By using Table 4-6 in conjunction with Table 4-7, the expected weight, power and volume characteristics of the I/O and bus control unit and the configuration assignment unit can be approximated for extended and new technology.

REFERENCES

C-1. "Computer/Converter Subsystem Study," Volume One, IBM No. 3-001057, April 1968.

C-2. Technical Memorandum, "Preliminary Memory Trade Studies," IBM 70-290-009, April 15, 1970.

C-3. "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974," MSFC-DRL-008. Line Item No. 161, IBM No. 70-238-0001, 14 January 1970.

C-4. "A Two-Cable Transformer Coupled Bus System," IBM No. 69-K44-0009, 15 October 1969.

C-5. "Preliminary Design of the Computation System for Shuttle Vehicle," 10 July 1970, Computer Division, Astrionics Laboratory, MSFC.

C-6. "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974," IBM No. 69-K44-0006F. Appendix C – Space Tug Astrionic System Study, Interim Report.

C-7. "Design Techniques for Modular Architecture for Reliable Computer Systems, NAS 8-24883, IBM No. 70-208-0002, 26 March 1970.

APPENDIX D

SPACE TUG NAVIGATION ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

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1.0 INTRODUCTION

This document presents the results of an analysis performed to: (1) relate mission characteristics to the space tug navigation and guidance requirements, (2) identify the candidate hardware which satisfy the established requirements, (3) define the mounting constraints of candidate hardware and (4) describe the technology trends for the candidate navigation sensors.

2.0 STUDY GUIDELINE AND GROUND RULES

The navigation analysis is based on the following ground rules and guidelines.

1. The tug will be primarily manned, but will be capable of automatic or remote control.

2. Autonomy is a design goal.

3. Automatic landing on the lunar surface will be at a prepared site, only.

4. The target vehicle for rendezvous and docking will be cooperative.

5. The navigation accuracy required is basically the same as for the LM.

6. Each mission is assumed to begin with all of the stages in the stacked configuration.

7. The accuracy requirements for the early flight phases (inertial N&G mode) is set depending on available radar range to effect a transition from the early N&G mode to a terminal guidance mode.

3.0 SUMMARY OF RESULTS

3.1 NAVIGATION REQUIREMENTS

Navigation and guidance requirements are first related to characteristics of the design missions established for the definition of the space tug astrionic system.

A method is discussed that permits intraorbit operations in a near autonomous mode. This method requires knowledge of either the target or chaser vehicle orbit and radar tracking of the target by the chaser vehicle. However, considerable time (about 18 hrs.) is needed due to the long synodic period for low earth orbits (100 NM and 300 NM orbits) when using Hohmann transfers.

The preliminary N&G accuracy requirements for orbit transfers are summarized in Table 3-1 and were obtained by adapting a set of Keplerian subroutines to determine range at nominal transfer completion as a function of errors at injection.

Table 3-1. Preliminary Initial Position and Measured Velocity Controlling Errors

TRANSFER FROM 100 NM. LOW EARTH ORBIT TO	WITH OFFSET AIM POINT												WITHOUT OFFSET AIM POINT											
	75 NM RADAR						300 NM RADAR						75 NM RADAR						300 NM RADAR					
	POSITION KM			VELOCITY M/S			POSITION KM			VELOCITY M/S			POSITION KM			VELOCITY M/S			POSITION KM			VELOCITY M/S		
	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$
300 NM ORBIT		1.0		1.1				3.9		4.2				2.0		2.2				7.8		8.4		
SYNCHRONOUS ORBIT-DIRECT ASCENT, NO PHASING	0.5				0.4		2.0				1.7		1.0				0.8		4.0				3.4	
SYNCHRONOUS ORBIT-DIRECT ASCENT, WORSE CASE PHASING	.07				.07		.25				.25		.13				.13		.51				.51	
SYNCHRONOUS ORBIT-LOW ALTITUDE PHASING ORBIT, NO UPDATE	.05				.05		0.2				0.2		0.1				0.1		0.4				0.4	
SYNCHRONOUS ORBIT-LOW ALTITUDE PHASING, POSITION UPDATE AT PHASING ORBIT PERIGEE	.25				.25		1.0				1.0		0.5				0.5		2.0				2.0	

From the study conducted and with no excess ΔV available, the best trajectory for achieving equatorial synchronous orbit with any desired longitude is the use of a low altitude phasing orbit. Four approaches were evaluated and gross results are shown in Table 3-2. Details are discussed in Section 4.0.

For the late midcourse correction for the earth to moon transfer, onboard sensors are required to measure vehicle to moon line-of-sight. Range and range rate data is desired. These studies showed that measurement of these parameters within 2% would provide a resultant orbit with perilune within radar range (300 NM) of the lunar space station (after any necessary phasing).

Follow-on in-depth analysis and simulations are required to:

- For the low earth orbit operations, perform a ΔV versus mission time study using the near autonomous N&G mode.
- Determine optimum degree of autonomy for space tug operations.
- Include second order effects in determining N&G accuracy requirements and update or correct accuracies shown.

Table 3-2. Synchronous Orbit Trajectory Comparison Matrix

TRAJECTORY	ADVANTAGES	DISADVANTAGES
DIRECT ASCENT WITH SLOW DRIFT TO STATION	• NOMINAL TRANSFER TIME. ($\simeq$5.3 HOURS) • NOMINAL N&G ACCURACY REQUIRED IF TARGET IS NEAR OPTIMUM LONGITUDE. • USES NOMINAL ΔV.	• LONG PHASING PERIOD ($\simeq$2 DAYS) AT NEAR SYNCHRONOUS ALTITUDE TO ARRIVE AT ANY REQUIRED LONGITUDE. • LONG PHASING PERIOD IMPOSES STRINGENT N&G REQUIREMENT OR NAVIGATION UPDATE.
LOW PHASING ORBIT	• MINIMUM TOTAL MISSION TIME FOR WORSE CASE TARGET LONGITUDE AND USING NOMINAL ΔV.	• STRINGENT N&G ACCURACY REQUIREMENTS WITHOUT POSITION UPDATE.
HIGH PHASING ORBIT	• USES NOMINAL ΔV.	• LONG TRANSFER TIME. (18-20 HRS.) • STRINGENT N&G ACCURACY REQUIREMENTS WITHOUT NAVIGATION UPDATE.
OVERSHOOT PHASING	• MINIMUM TOTAL MISSION TIME. • NOMINAL N&G ACCURACY REQUIRED.	• EXCESS ΔV REQUIRED. (690 M/S WORST CASE)

- Show need and cost of the conservatism included in the use of an offset aim point.
- Perform the radar range versus inertial guidance accuracy requirements trade study, including cost effectiveness, for all representative missions.
- Develop and provide an error budget for the N&G hardware.

3.2 NAVIGATION SENSOR SELECTION

Figure 3-1 depicts the preliminary choice of sensors to perform the navigation function for the spectrum of space tug missions. It should be noted that the configuration chosen is based on limited trade and component data and does not necessarily reflect the final configuration for space tug navigation.

The layout is configured for the manned synchronous orbit mission which includes automatic rendezvous and docking. The navigation subsystem is comprised of the following units.

(1) IMU (Strap-down)

(2) Star trackers (2)

(3) Rendezvous and Docking Radar (Laser Radar)

(4) Landmark tracker

(5) Horizon Sensor

For missions that include a lunar landing as one of the mission phases, a landing radar is required.

A matrix of navigation sensors versus tug missions is presented in Figure 3-2. The estimated size, weight, and power requirements for each subsystem are given in Table 3-3. The functions of each unit and the mission phases where they are employed are presented below and summarized in the matrix of figure 3-3.

3.2.1 IMU

The IMU is employed for all active mission phases to measure vehicle thrust acceleration and to provide vehicle attitude reference. A strap-down configuration was chosen since it will perform to the required accuracy and offers advantages in redundancy, cost, weight, and power over four redundant conventional gimballed IMU's, as well as an advantage in inherent reliability. However, a tradeoff comparison will be made in greater depth in follow-on studies. The potential advantages which must be evaluated for the gimballed configurations include:

- Greater maturity and production base which should result in lower development costs.
- Capability for launch pad calibrations which eliminates dependence on long-term stability of inertial sensor error coefficients.

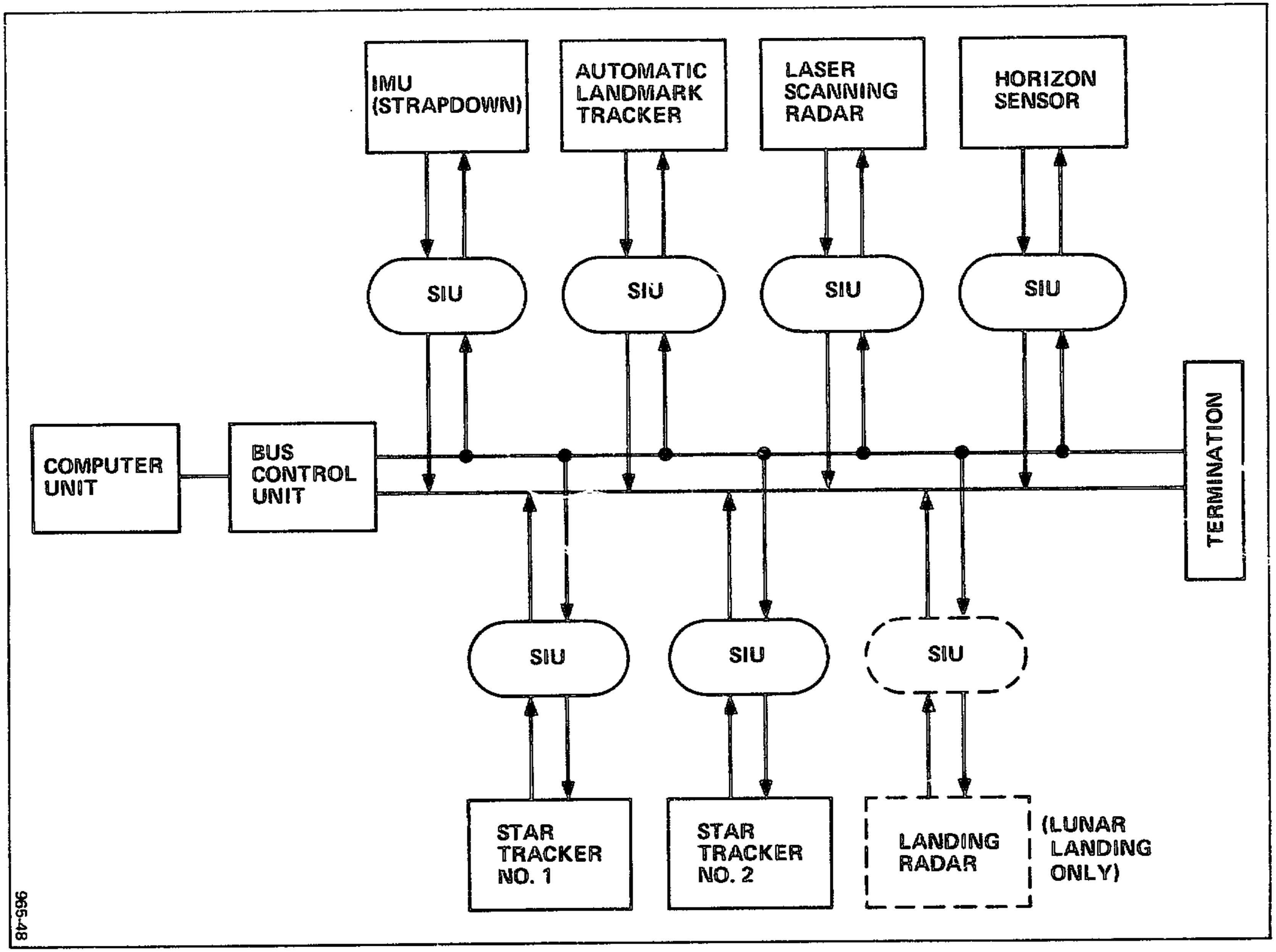

Figure 3-1. Space Tug Navigation Layout

EQUIPMENT \ MISSIONS	REUSABLE FIRST STAGE SYNC	REUSABLE SECOND STAGE SYNC	EXPENDABLE SYNC	EARTH ORBIT OPS	LUNAR ORBIT OPS	PLANITARY FIRST STAGE	PLANETARY SECOND STAGE	RNS	FOUR STAGE SATURN V	LUNAR LANDING
IMU STRAPDOWN	●	●	●	●	●	●	●	●	●	●
LASER RADAR	●	●	●	●	●	●		●	●	●
STAR TRACKERS (2)	●	●	●	●	●	●	●	●	●	●
LANDMARK TRACKER	●	●	●	●	●	●		●	●	●
HORIZON SENSOR		●	●			●		●	●	
LANDING RADAR										●

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Figure 3-2. Matrix Relating Navigation Equipment To Space Tug Missions

Table 3-3. Navigation Sensors Physical Characteristics

SUBSYSTEM	WEIGHT (LBS)	POWER (WATTS)	VOLUME (CU. FT.)	EST MTBF (HRS.)	EST SERVICE LIFE
IMU*	80	200	0.6	9,000	3,000 HRS.
LASER RADAR**	28	30	1.16	UNKNOWN	UNKNOWN
STAR TRACKER (2)	23 EA.	20 AVE EA. 30 PK	0.95 EA.	20,000	7 YEARS
LANDING RADAR	34	80	1.10	5,000	
LANDMARK TRACKER	30	40	1.0	40,000	2,000 HRS.
HORIZON SENSOR	17	10 AVE 13 PK	0.18	174,000	7 YEARS

* HEXAD STRAPDOWN UNIT
** CHASER VEHICLE EQUIPMENT

3.2.2 Star Trackers, Landmark Tracker, and Horizon Sensor

Two star trackers were selected for automatic IMU alignment. The star trackers along with either a landmark tracker or horizon sensor was selected to perform autonomous vehicle state updates. The landmark tracker is used for distances up to a maximum of 10,000 NM from a celestial body. Navigation beyond an altitude of 10,000 nmi will require a horizon sensor to reliably, automatically, and accurately update the vehicle state vector during synchronous orbit, planetary, and translunar transearth missions. These units will be used for navigation initializations and orbital-lunar surface navigation.

The two star trackers and the landmark tracker or horizon sensor serve the same functions the telescope/sextant provide for the LEM. For manned operations, the telescope/sextant is sufficient. However, for unamnned missions, the star trackers and automatic landmark tracker or horizon sensor are required.

3.2.3 Rendezvous and Docking Radar

The function of the rendezvous and docking radar is to provide range, range rate, angular position, and angular rate with respect to a target vehicle or location.

The laser radar was selected for this function since it can be used for rendezvous, docking and automatic landing. This radar has an advantage over the present microwave rendezvous radars in that it provides the accuracy required for automatic docking as well as for rendezvous. Also, it can be used as a landing aid for unmanned lunar landing at a prepared site.

At present, a laser scanning radar developed by ITT in cooperation with NASA MSFC has performed successful simulated dockings at the Martin Denver Simulation Laboratory. A similar laser radar was mounted on a helicopter and successfully docked with a docking collar placed atop of a truck on the side of a mountain. The present state of development of this laser indicates an availability in the 1973-1978 time frame.

NAVIGATION SENSORS \ FUNCTION	IMU ALIGNMENT	NAV. UPDATE	RENDEZVOUS	DOCKING	LUNAR DESCENT AND ASCENT	TRANSFER BURNS	COAST	SENSOR OUTPUT DATA
IMU		●	●	●	●	●	●	Δ ANGLE AND Δ VELOCITY
LASER RADAR			●	●	●			RANGE, RANGE RATE, TARGET ANGLE DATA
STAR TRACKERS (2)	●	●						STAR ANGLE DATA
*LANDMARK TRACKER		●						LANDMARK ANGLE DATA
*HORIZON SENSOR		●						HORIZON ANGLE DATA
LANDING RADAR					●			ALTITUDE AND VELOCITY

*LANDMARK TRACKER USED FOR ALTITUDES LESS THAN 10 K NM

*HORIZON SENSOR USED FOR ALTITUDES GREATER THAN 10 K NM

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Figure 3-3. Required Navigation Sensors and Their Functions

3.2.4 Landing Radar

The function of the landing radar is to provide velocity and altitude with respect to the lunar surface, and prior to acquisition of the landing site by the laser radar, to supplement the IMU inertially measured quantities during lunar landing.

The landing radar presently employed on the LM appears to provide the functions and accuracy required for the space tug.

4.0 MISSION CHARACTERISTICS AND NAVIGATION AND GUIDANCE REQUIREMENTS

Figure 4-1 lists the space tug missions (with all vehicles) and the mission phases expected for the spectrum of space tug design missions. The commonality of the mission phases for the missions indicates that there is also much commonality for the functional requirements. Some pertinent mission characteristics and N&G requirements are discussed in this section.

Requirements and specifications must be related to the functional hardware to be used. Thus, the assumption made in this requirements analysis is the utilization of a core G&N subsystem consisting of an inertial measurement unit (IMU), digital computer with needed flight program (software), and a target sensor (radar). This core system will operate in two primary modes:

- Inertial Guidance – for preprogrammed trajectories.
- Terminal Guidance – for accurate terminal phase guidance requiring direct target data.

Supporting functions and equipment required for this core system include the following:

- IMU Alignment – The duration of all seven design missions is in excess of several hours and thus onboard sensor(s) are required for periodic IMU alignment.
- Space Navigation – Providing onboard navigation to minimize ground station support is a design goal. With the IMU alignment requirement (sensors available), the hardware impact in providing position data is not considered to be large. Software impact is also not excessive.
- Trajectory Computations – The capability and data needed to compute onboard precision trajectories is mission and mission phase dependent.

A review of the missions, past studies, and other available data indicates that:

(1) The intraorbit transfers (earth and lunar orbit operations design missions) are the most likely candidate for autonomous operation.

(2) The synchronous orbit mission will impose the most stringent N&G requirements.

MISSION PHASES \ MISSION	SYNCHRONOUS (EXPENDABLE TUG)	SYNCHRONOUS (REUSABLE - 1ST TUG)	SYNCHRONOUS (REUSABLE - 2ND TUG)	LUNAR LANDING	EARTH ORBIT OPERATIONS	LUNAR ORBIT OPERATIONS	PLANETARY (REUSABLE 1ST TUG)	PLANETARY (EXPENDABLE 2ND TUG)	REUSABLE NUCLEAR SHUTTLE	FOUR STAGE SATURN V
BOOST TO LOW EARTH ORBIT										●
INTER-ORBIT TRANSFER	●	●	●	●	●	●	●	●	●	●
RENDEZVOUS	●	●	●	●	●	●	●		●	●
DOCKING/ UNDOCKING	●	●	●	●	●	●	●		●	
COAST	●	●	●	●	●	●	●	●	●	●
ACTIVATION/ DEACTIVATION	●	●	●	●	●	●	●	●	●	TBD
MAINTENANCE		●	●	●	●	●	●		●	●
FUELING CONTROL	●	●	●	●	●	●	●	●	●	
STORAGE		●	●	●	●	●	●		●	
LUNAR DESCENT AND ASCENT				●						

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Figure 4-1. Space Tug Mission Phase Commonality

(3) The late midcourse correction for the earth to moon transfer, and for automated missions, will impose special sensor and/or operational requirements.

This analysis, due to the compacted schedule, was directed toward these three areas.

In determining the onboard N&G system accuracy requirements, several constraints must be considered, including availability and limitations of the ground stations, number of trajectory correction burns, and fuel allocated. The approach taken is to set the N&G accuracy requirements for the inertial guidance phase as needed to achieve radar lock-on for initiating the terminal guidance phase. Also a design goal was set to minimize ground support requirements.

4.1 INTRAORBIT TRANSFERS

Orbital transfers/operations are to be performed in both low earth orbit and lunar orbit. The representative mission profile for tug orbital operations consist of the tug executing a Hohmann transfer from the space station orbit to the target orbit, rendezvous and dock, load payload, transfer to the space station orbit, and rendezvous and dock to the space station. In performing these transfer maneuvers, there are two basic approaches depending on the difference in the tug and target altitude (Δh) and the guidance hardware used.

(a) If Δh is large, an inertial guidance phase plus a terminal guidance phase is required. For the inertial guidance, inputs to the tug guidance system are required to define both the tug and target state vectors either from ground track or from the space station.

(b) If Δh is small, i.e., target is within tug radar range, only a terminal guidance phase is required with inputs from the space station defining the tugs' initial state vector.

Approach (b) above, provides considerable less reliance on data from ground stations and should provide more flexible lunar orbit operations. Description of this terminal guidance approach is as follows.

If the space station (target) is in a circular near-earth orbit, and the space tug is nearby, the motion of the tug with respect to the station can be defined simply by introducing a rotating coordinate system with the origin at the station. With reference to Figure 4-2, let:

X be in the direction of motion of the station

Y be in the local vertical (up at station)

Z be in the negative orbit normal at the station

then, X, Y and Z are coordinates of the tug and the equations of motion are:

$$\ddot{X} - 2\omega\dot{Y} = 0$$

$$\ddot{Y} + 2\omega\dot{X} - 3\omega^2 Y = 0$$

$$\ddot{Z} + \omega^2 \dot{Z} = 0$$

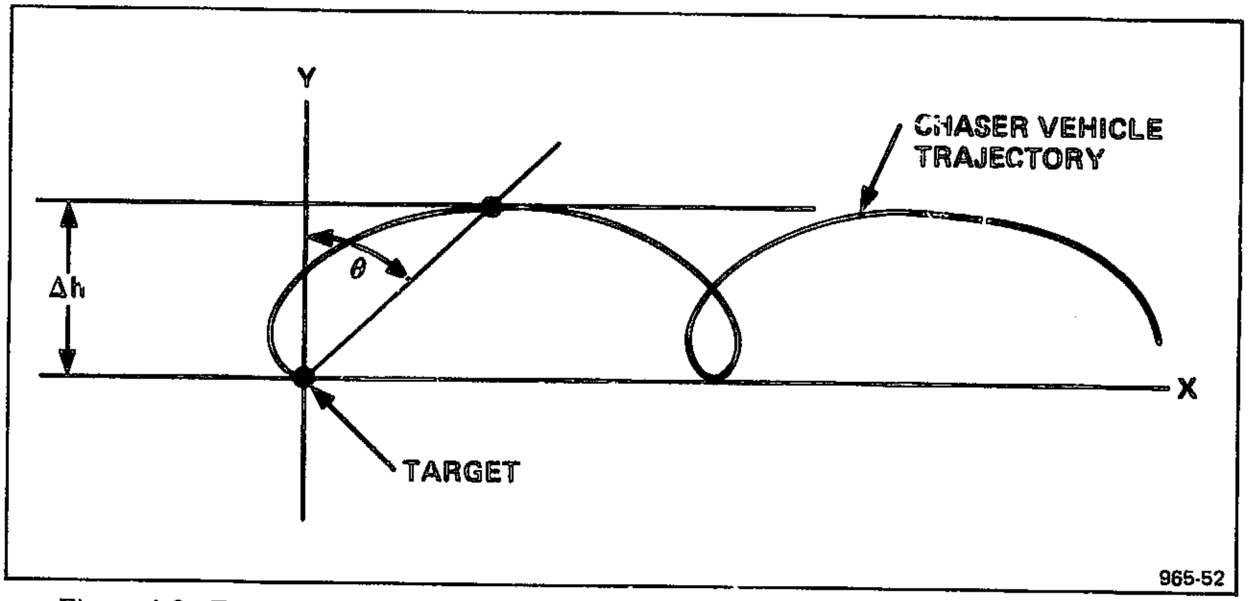

Figure 4-2. Transfer Trajectory In Rotating Coordinate System

These equations have the solution:

$$X = 2A \sin(\omega t + \phi) + a + 3b\omega t$$

$$Y = A \cos(\omega t + \phi) + 2b$$

$$Z = B \sin(\omega t + \phi_Z)$$

where:

$$A = \frac{1}{\omega} \left[\dot{Y}_o^2 + (2\dot{X}_o - 3\omega Y_o)^2 \right]^{1/2}$$

$$B = \left[Z_o^2 + \left(\frac{\dot{Z}_o}{\omega} \right)^2 \right]^{1/2}$$

$$a = X_o + \frac{2}{\omega} Y_o$$

$$b = 2Y_o - \frac{\dot{X}_o}{\omega}$$

$$\phi = \tan^{-1} \left(- \frac{\dot{Y}_o}{3\omega Y_o - 2\dot{X}_o} \right)$$

$$\phi_Z = \tan^{-1} \left(\frac{\omega Z_o}{\dot{Z}_o} \right)$$

These solutions permit the orbital transfer operations to be performed with:

(a) Definition of either (but not both) the target or tug state vector and

(b) The following equipment onboard the tug:

(1) Platform aligned (permitting location of the local vertical) or an unaligned platform with a horizon or other sensor(s) to determine the local vertical

(2) Radar or optical tracker to provide range, range rate, and line-of-sight angles in tug reference.

As an illustration, suppose that the tug is in a circular orbit coplanar with the station, but at an altitude, Δh, above the station.

Then an optimum transfer is made by using impulses in the X direction only. The equations of motion for arriving at the space station with velocity $\dot{X}_o$ are

$$X = \frac{4}{\omega} \dot{X}_o \sin \omega t - 3 \dot{X}_o t$$

$$Y = \frac{-2\dot{X}_o}{\omega} (\cos \omega t - 1)$$

where t is zero at the conclusion of the maneuver.

The minimum energy transfer is made by waiting (phasing) until the tug lies ahead of the station at an angle from the vertical

$$\theta = 79^{\circ} 36' = \tan^{-1} (3\pi - 4)$$

then applying impulse to give delta velocity

$$\Delta V = \frac{\Delta h \omega}{2}$$

Approximately the same ΔV must be applied at the origin to complete the transfer maneuver. These equations define a Hohmann transfer in the rotating coordinate system.

For orbital operations using an inertial guidance system for the orbit transfer maneuvers and then a terminal guidance system for rendezvous, the inertial guidance system must provide sufficient accuracy so that the target is within radar or optical range at completion of the transfer. A set of Keplerian subroutines were used to determine range at nominal transfer completion as a function of error at injection. While suitable for determining error, these routines do not include second order effects which would have to be taken into account to determine a nominal trajectory (e.g., earth oblateness).

Figure 4-3 shows the geometry of a Hohmann transfer, the coordinate system used, and use of an offset aiming point. The duration of the transfer is determined from the nominal. For the perturbed cases, the time after perigee is determined from the initial conditions, and the nominal duration added to determine the time at the end of the

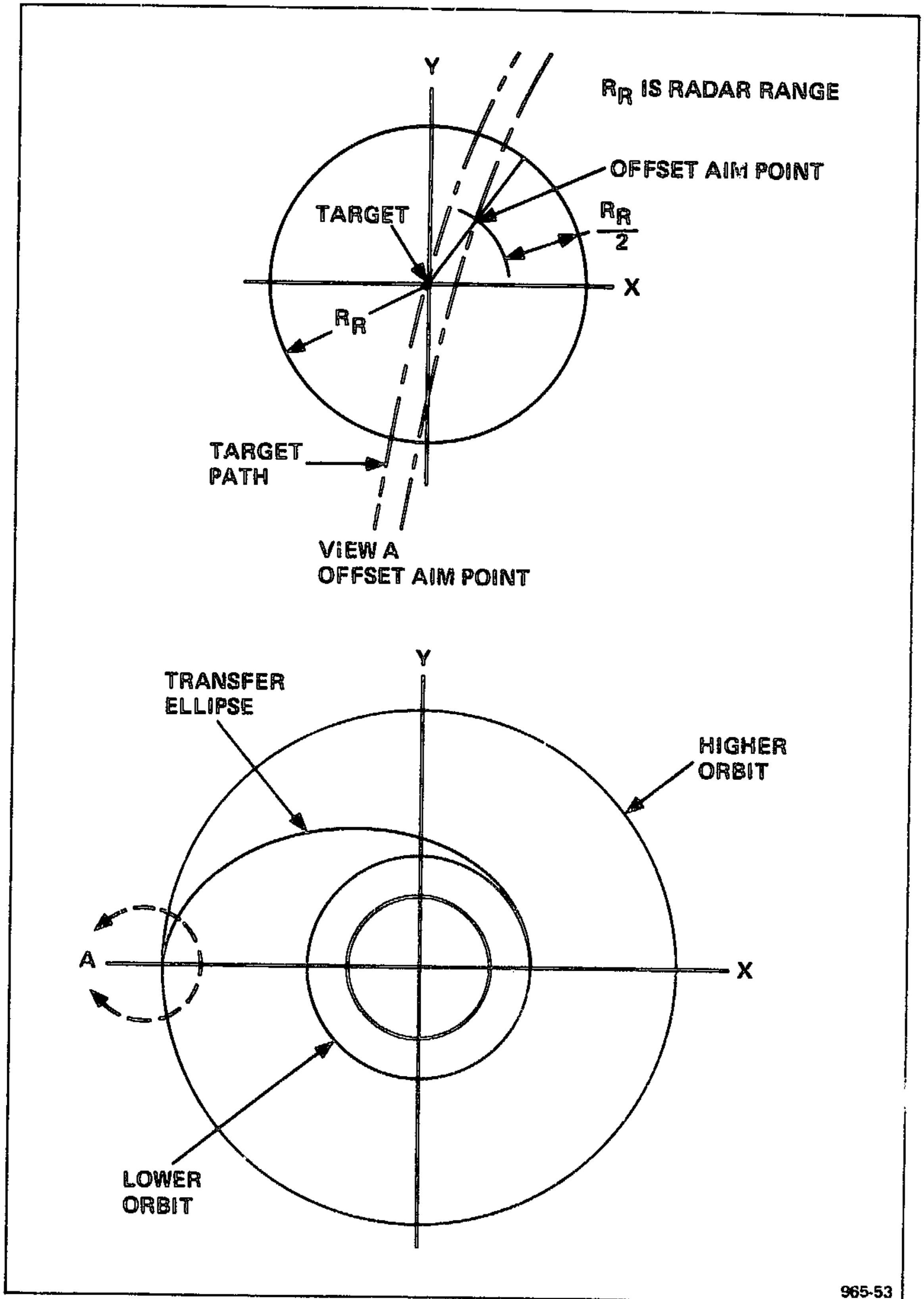

Figure 4-3. Hohmann Transfer Geometry

perturbed maneuver. The errors given are therefore the error at expected completion time and not necessarily the error at closest approach. The offset aiming point is used to provide a favorable position for the rendezvous phase and to limit the volume of space the radar must search.

Table 4-1 shows the errors for two cases. The miss distances are proportional to the input errors. For systems with a radar range of 550 km (≃300 NM) using the offset aiming point as shown in Figure 4-3, the error in X cannot exceed 4.2 m/s and the error in Y cannot exceed 3.9 km. For a system with a radar range of 140 km (≃75 NM), the error in X cannot exceed 1.1 m/s and error in Y cannot exceed 1.0 km. These errors were obtained by assigning one-half the allowable miss distance to initial position errors and the remaining half to velocity measurement errors. Other combinations may be used.

4.2 SYNCHRONOUS ORBIT MISSION

The equatorial synchronous orbit mission, as shown in Figure 4-4 can be performed using only two burns, provided the phasing can be completed by placing the tug into a slightly lower orbit and permitting it to drift into the required position (longitude).

The Hohmann transfer is initated when the tug crosses the equatorial plane. By delaying the initiation of the transfer for a few orbits, the synchronous orbit position can be brought within one orbit period (about 90 minutes) of the injection point. Table 4-2 shows the time for the tug to "catch up" for various altitudes below synchronous orbit.

Table 4-1. Hohmann Transfer From 100 NM Earth Orbit To 300 NM Orbit

INPUT ERROR	MISS DISTANCE
$\dot{X}$, 5 M/S	164. KM
$\dot{Y}$, 5 M/S	42. KM
$\dot{Z}$, 5 M/S	.03 KM
X, 1 KM	10. KM
Y, 1 KM	38. KM
Z, 1 KM	1. KM

INPUT ERROR	MISS DISTANCE
$\dot{X}$, 2 M/S	75 KM
$\dot{Y}$, 2 M/S	17 KM
$\dot{Z}$, 2 M/S	NEGLIGIBLE
X, 0.5 KM	5 KM
Y, 0.5 KM	19 KM
Z, 0.5 KM	0.5 KM

Table 4-2. Altitude Versus Catch Up Time

ΔH (NM)	$\Delta\omega$ (10^{-6} RAD/SEC)	T (DAYS)
50	.2402	18.5
63	.3030	14.7
100	.4819	9.2
200	.9693	4.6
300	1.4621	3.0
400	1.9603	2.27
500	2.4642	1.81

where,

Δh = distance catch up orbit is below synchronous orbit

$\Delta\omega$ = difference in orbital rates

t = time to catch up

The phasing operation can be performed much more quickly using a phasing orbit, but additional requirements are placed on the N&G system.

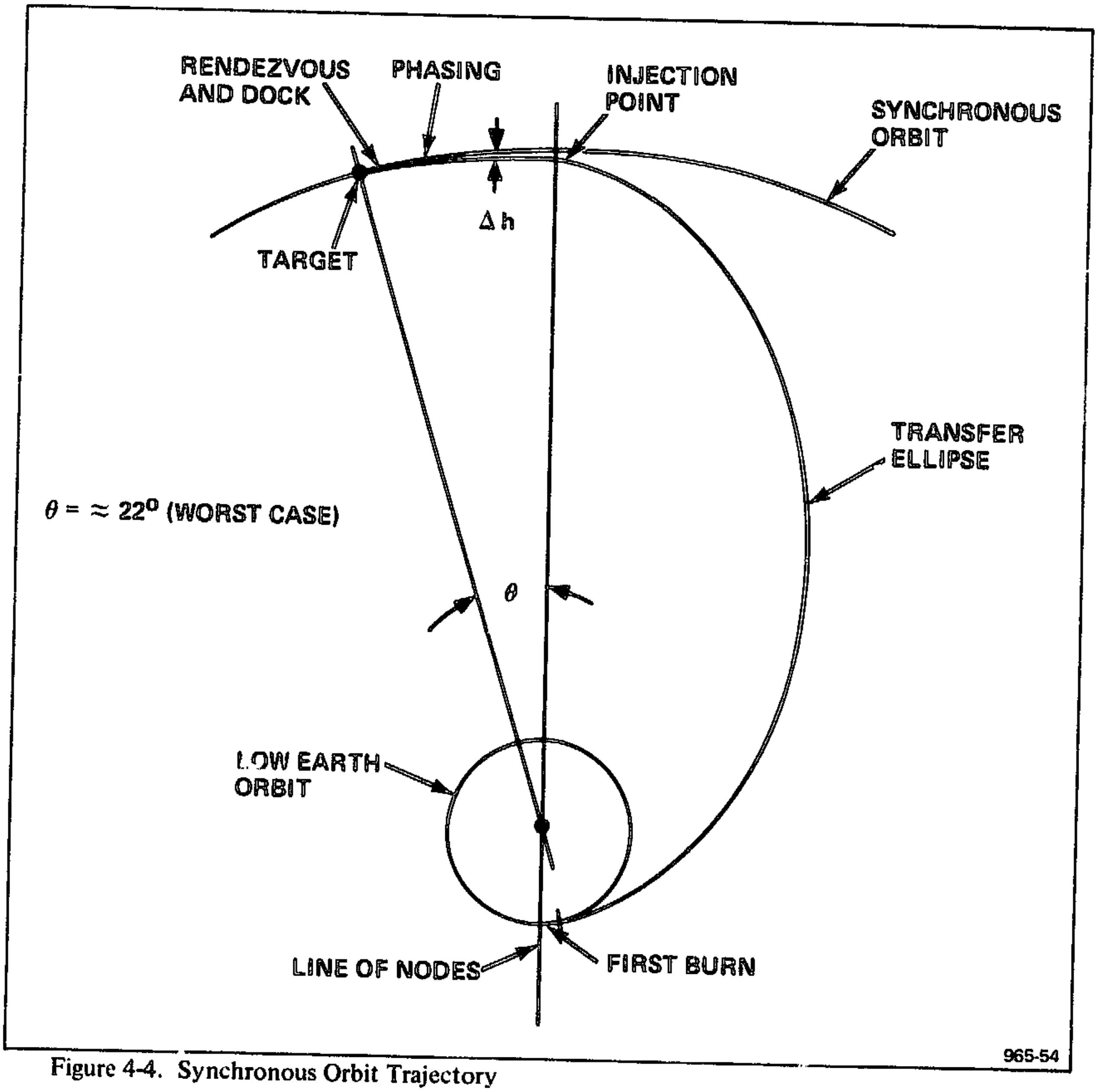

Figure 4-4. Synchronous Orbit Trajectory

The displacement due to unit N&G errors for the transfer, using the same approach as described earlier in section 4.1 for orbital transfers, are shown in Table 4-3.

For systems with a radar range of 550 km (≃300 NM) and using the offset aiming point as shown in Figure 4-3, the error in $\dot{Y}$ cannot exceed 1.7 m/s and the error in X cannot exceed 2 km. For a system with a radar range of 140 km (≃75 NM), the error in $\dot{Y}$ cannot exceed 0.4 m/s and in X cannot exceed 0.5 km. The errors were obtained by allowing one-half the allowable miss distance to initial position errors and the remaining half to velocity measurement errors.

For drift phasing after obtain equatorial synchronous orbit, it is required that the variation in Y never exceed the radar range for autonomous operation. From the equations derived in section 4.1,

$$Y_{max} = A + 2b$$

where,

$$A = \frac{1}{\omega} [\dot{Y}_o^2 + (2\dot{X}_o - 3\omega Y_o)^2]^{1/2}$$

$$2b = 4Y_o - \left(\frac{2\dot{X}_o}{\omega}\right)$$

$$\omega = \frac{2\pi}{(24)(3600)} = 0.728 \times 10^{\text{[illegible]}} \text{ rad/sec}$$

Note: X and Y are interchanged.

Table 4-3. Hohmann Transfer From 100 NM To Synchronous Orbit

INPUT ERROR	RESULTANT ERROR AT APOGEE						MISS DISTANCE
	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$	KM (RSS)
$\dot{X}$, 1 M/S	0.	9.7	0.	0.36	0.	0.	9.7
$\dot{Y}$, 1 M/S	61.1	56.9	0.	7.97	2.16	0.	83.2
$\dot{Z}$, 1 M/S	0.	0.	C.	0.	0.	0.15	0.15
X, 1 KM	54.2	51.3	0.	7.2	1.8	0.	74.0
Y, 1 KM	0.	8.8	0.	.33	0.	0.	8.8
Z, 1 KM	0.	0.	6.4	0.	0.	0.	6.4

For the unit errors from Table 4-3,

$Y_O = 115.3$ km

$\dot{X}_O = 3.96$ m/s

$\dot{Y}_O = 15.86$ m/s

$y_{max} = 1075$ km

Also,

$$Y_{max} = R - H$$

where,

R is the radar range

H is nominal altitude difference for phasing

Then by proportion, the allowable guidance error versus phasing time is shown in Table 4-4.

Table 4-4. Guidance Error Versus Phasing Time

PHASING TIME (DAYS)	MAXIMUM GUIDANCE ERROR 300 NM RADAR		MAXIMUM GUIDANCE ERROR 75 NM RADAR	
∞	0.51 KM AND 0.51 M/S		0.13 KM AND 0.13 M/S	
18.5	0.42	0.42	0.04	0.04
14.7	0.40	0.40	0.02	0.02
9.2	0.34	0.34	NOT POSSIBLE	
4.6	0.17	0.17	NOT POSSIBLE	
3.0	0.0	0.0	NOT POSSIBLE	
	SHORTER TIMES NOT POSSIBLE			

In Table 4-4 the position and velocity errors were, again, arbitrarily taken as equal. Also, some error should be allowed for the planned difference in orbital altitude, but this error is small.

Several methods are available for phasing, i.e., in reaching the required longitude in an equatorial synchronous orbit, starting from a low earth orbit. These include:

(1) Slow Drift – Injection into a orbit of slightly less semi-axis major than desired as discussed in the preceding paragraphs and illustrated in Figure 4-4. The impulse required to end the drift is small and can be supplied by small thrusters.

(2) Low Phasing Orbit – At the beginning of the Hohmann transfer, only part of the transfer impulse is applied. This results in a phasing orbit with apogee dependent on the longitude of the target hover point and with perigee at the low earth orbit. The transfer is begun after one phasing orbit. This method is illustrated in Figure 4-5.

(3) High Phasing Orbit – At the end of the Hohmann transfer orbit, as illustrated in Figure 4-6, only part impulse to complete injection is applied. The result is a phasing orbit with apogee at synchronous orbit and whose period is dependent on the longitude of the target hover point. Injection is completed after one phasing orbit.

(4) Overshoot Phasing – An excess of impulse is applied at the beginning of the transfer. Synchronous altitude is then reached more quickly, even through the beginning of transfer must be delayed somewhat so that injection still takes place on the line of nodes. Injection takes place when synchronous altitude is reached (before apogee). This method is illustrated in Figure 4-7.

Phasing can be accomplished without impulse penalty in the second and third approaches. The low phasing orbit imposes less stringent requirement on the guidance and navigation system than the other approaches.

Table 4-5 shows the errors for the maximum required (low altitude) phasing orbit at the end of the nominal period. These are errors developed in an autonomous system and can be improved by direct observation of the phasing orbit period. If the errors in the second half of the divided burn are neglected, these errors can be translated into equivalent errors at synchronous orbit injection.

If the effective range of the radar is 550 km (300 NM), then errors at the end of the injection into the phasing orbit no larger than 0.2 km in position and 0.2 m/s in velocity are needed to insure acquisition of the target by radar (offset aim point included). If about half the time/position error can be removed prior to the transfer burn, these errors may be increased to 1 km in position and 1 m/s in velocity. For a radar with an effective range of 140 km (75 NM), and with half the errors removed prior to transfer burn and using the offset aim point, the errors should be no larger than 0.25 km in position and 0.25 m/s in velocity.

Examination of the error matrix (Table 4-3) shows that only the radius magnitude and total velocity are important i.e., X and $\dot{Y}$. These errors are small compared to the errors at transfer so that the neglect of the second portion of the injection is justified.

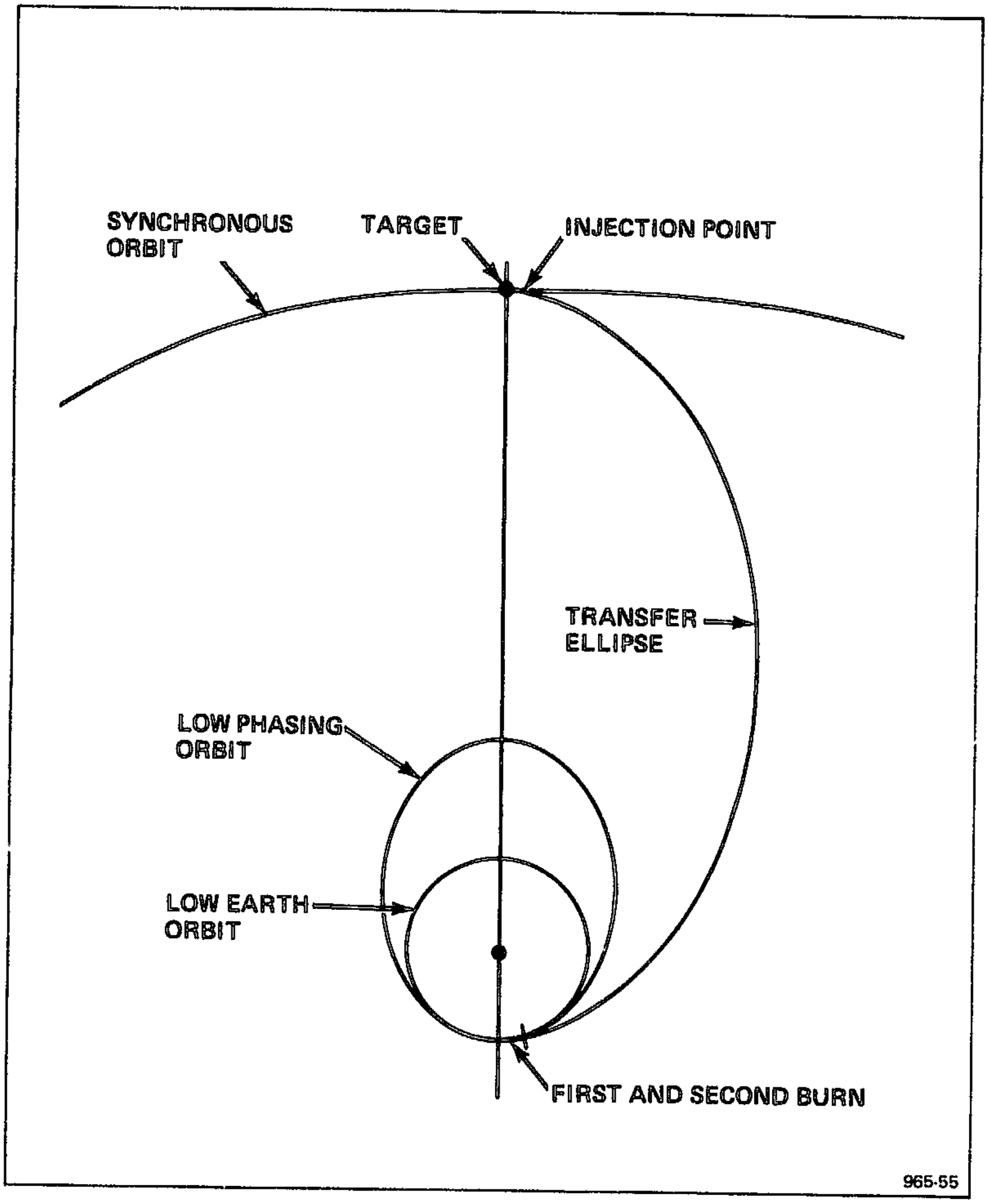

Figure 4-5. Synchronous Trajectory Using Low Altitude Phasing Orbit

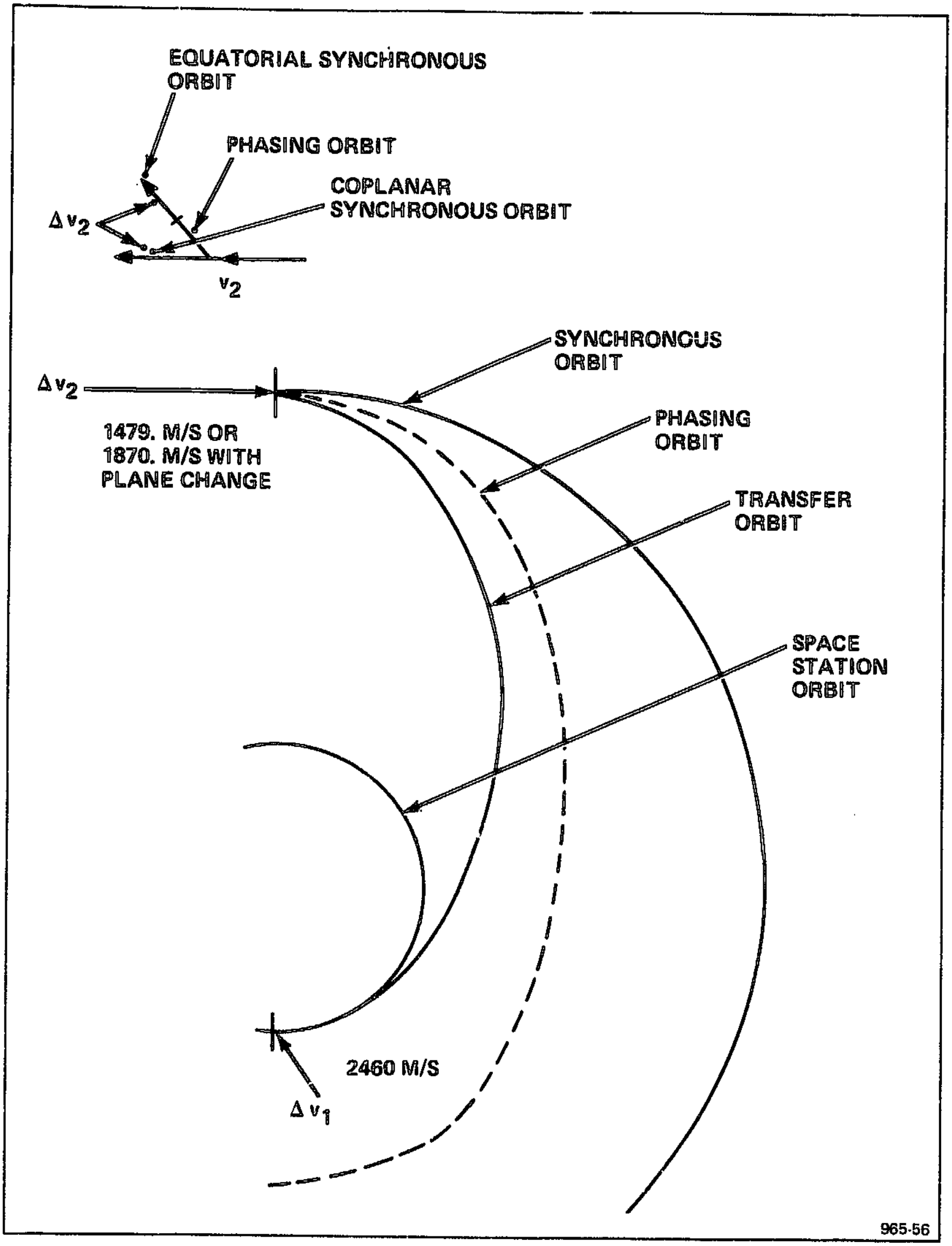

Figure 4-6. Synchronous Orbit Mission Using High Phasing Orbit Trajectory

Table 4-5. Low Altitude Phasing Orbit Error Matrix

UNIT ERRORS	X	Y	Z	$\dot{X}$	$\dot{Y}$	$\dot{Z}$	TOTAL	EFFECT AT SYNC ORBIT PER UNIT	MISS DISTANCE (KM)
ΔX	0.970	0.	0.	0.	0.265	0.	1.23	74	91.
ΔY	70.04	0.99	0.	.004	69.	0.	140.	8	1120.
ΔZ	0.	0.	1.	0.	0.	0.	1.	6	6.
$\Delta \dot{X}$	71.	0.	0.	1.	70.	0.003	142.	9.7	1380.
$\Delta \dot{Y}$	0.04	0.	0.	0.	0.63	0.	0.67	83.2	56.
$\Delta \dot{Z}$	0.	0.	0.	0.	0.	1.	1.	0.05	0.

NOTE: UNIT ERRORS ARE 1 KM IN POSITION AND 1 M/S IN VELOCITY

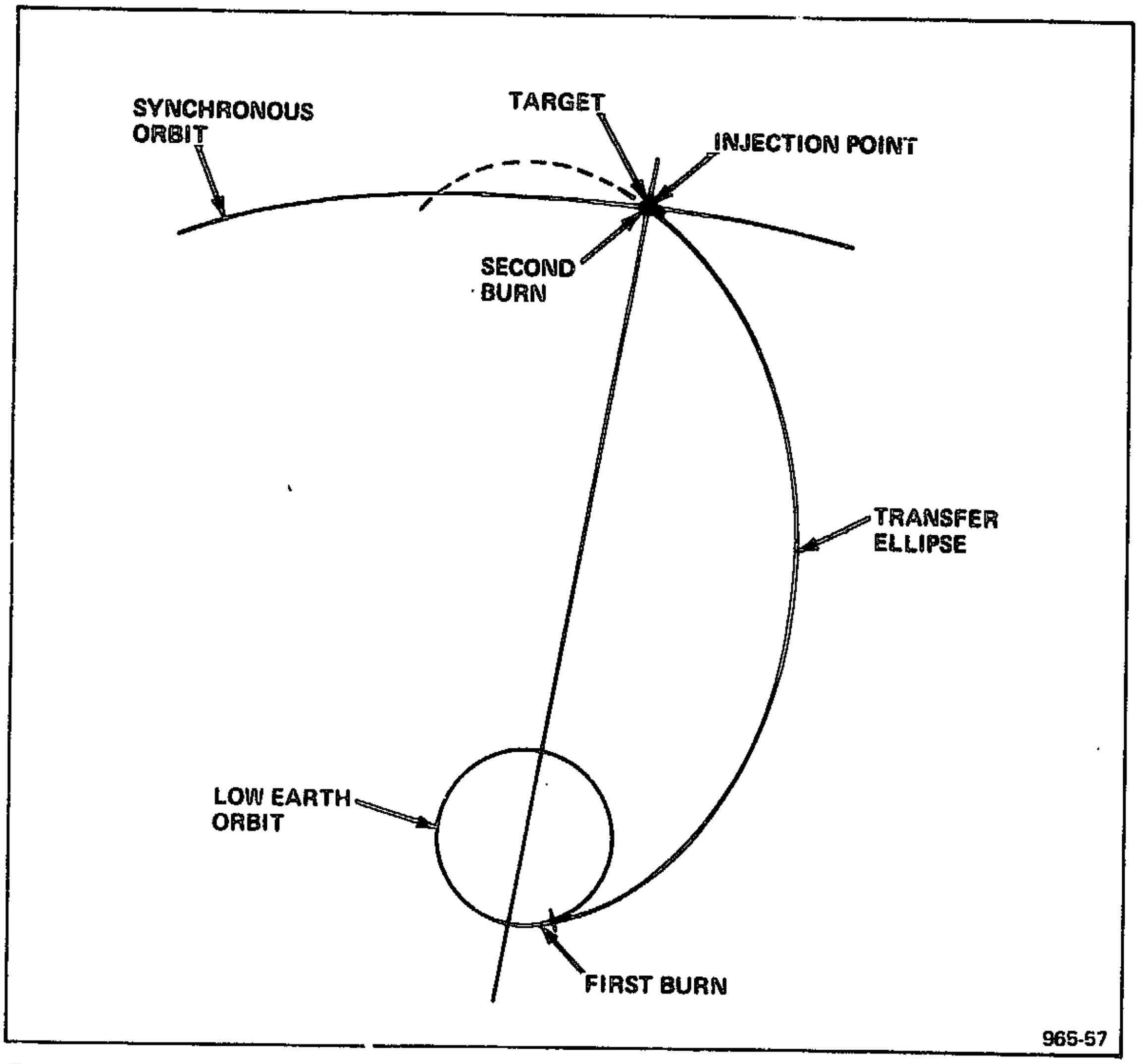

Figure 4-7. Synchronous Orbit Trajectory Using Overshoot

4.2.1 Position Update at Perigee

For the low altitude phasing orbit approach the onboard navigation requirements may be relaxed for synchronous orbit injection by using ground track or an onboard horizon sensor. If the error in measured position at the end of the phasing orbit were zero, the requirements would be the same as without the phasing orbit (i.e., about 3 m/s without offset aim point).

The phasing orbit returns through the same point, in the patched conic approximation, so that the only error is in locating the point at which the burn resumes. If the error is u km in Y, this causes also an error on the order of:

$$\frac{u}{r_p} \ (v_p) = 1.32 \text{ m/s in } \dot{X}$$

where

r_p = radius at perigee of phasing orbit, 6929 km

v_p = velocity at perigee of phasing orbit, 9121 m/s

Second order effects can be neglected. Then for a radar range of 550 km (300 NM), and a fraction k of the error:

$$k\ 550 < 8\ u + (9.7)\ (1.32\ \mu)$$

for k = 0.5, then u = 28 km.

It appears that a suitable system can be devised with a velocity error 1 m/s and a single tracking point near the end of the phasing orbit with a 3σ error of 28 km.

Another suitable system could have a velocity error of 1 m/s above, and a horizon sensor – star tracker system capable of determining local vertical to within 0.12 degrees. The re-ignition point is determined by duplication of the local vertical orientation in space at the beginning and the end of the phasing orbit.

4.2.2 Platform Alignment

For a 100 NM radar, such as a laser radar, and from the above discussion, a requirement exists for 0.33 m/s in velocity measurement with position update using either ground track or an onboard horizon sensor and a star tracker.

The velocity increment for injection into the transfer orbit is 2460 m/s. If the error in the forward direction is to be small, say 1/5 of the permitted error, then one alignment requirement is:

$$(0.33)\ (1/5) = 0.066 = 2460\ (1 - \cos e_1)$$

hence, e_1 = 0.42 degrees.

The most significant cross velocity component is that in X. If it is required that this component produce less than 10 km error at synchronous orbit arrival, then the error in X can be 10/9.7* = 1.03 m/s and a second error requirement is:

$$1.03 = 2460 \sin e_2$$

whence, e_2 = 0.024 degrees = 1.5 minutes of arc.

This last requirement is evidently the controlling error for platform alignment for synchronous orbit injection.

4.2.3 Overshoot ΔV Requirement

Figure 4-8 shows the additional impulse requirement for the overshoot phasing method of achieving a required longitude in equatorial synchronous orbit. From the figure, the impulse required without phasing is about 4330 m/s. With worse case phasing, which is a delay of one period of the low earth orbit, the impulse requirement becomes 5020 m/s. For comparison, both the high phasing orbit and the low phasing orbit require no additional impulse beyond the 4330 m/s, and the slow drift requires only slightly more.

4.2.4 Satellite Placement

The allowable simultaneous errors for an equatorial synchronous orbit satellite in order for the satellite to have a drift rate along its orbit smaller than 1°/yr. away from its desired station is as follows:

Inclination error	±1/2 deg of arc
Altitude error, deviation from ideal value	±107 meters
Maximum injection velocity magnitude error from circular speed	±6.7 m/sec
Maximum injection velocity direction error, from horizontal	7-1/2 min of arc

Achieving a synchronous orbit injection with this order of accuracies is considered to require a vernier guidance phase using ground track.

*From Table 4-5.

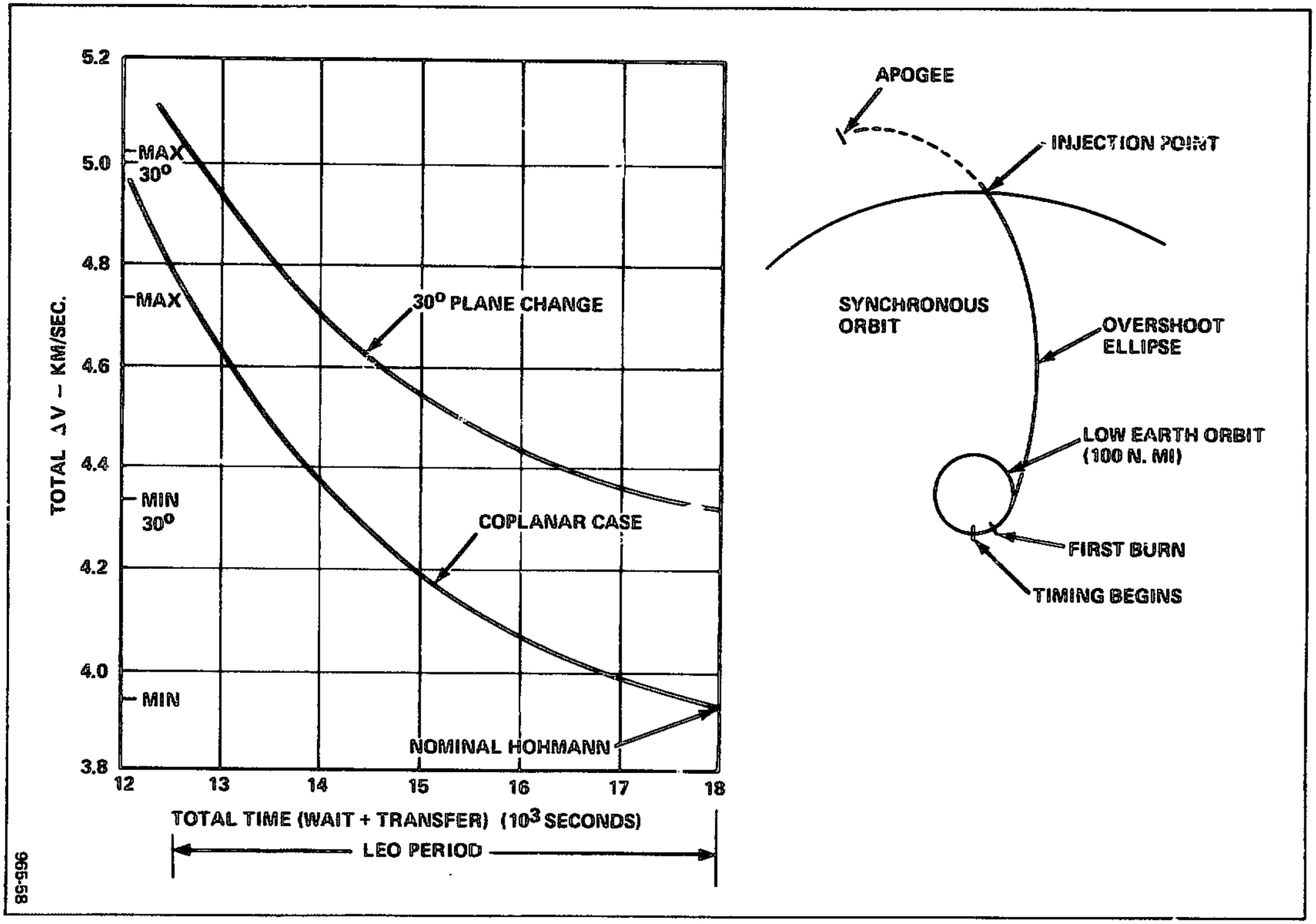

Figure 4-8. Overshoot Phasing ΔV Versus Hover Point

4.3 LUNAR MISSIONS

4.3.1 Lunar Orbit Injection

Injection into a lunar orbit cannot be made without some form of terminal guidance. If a sensor is used, as illustrated in Figure 4-9, which can determine lunar angular diameter and position against a star background or in inertial coordinates, a procedure for determining the final midcourse correction can be determined.

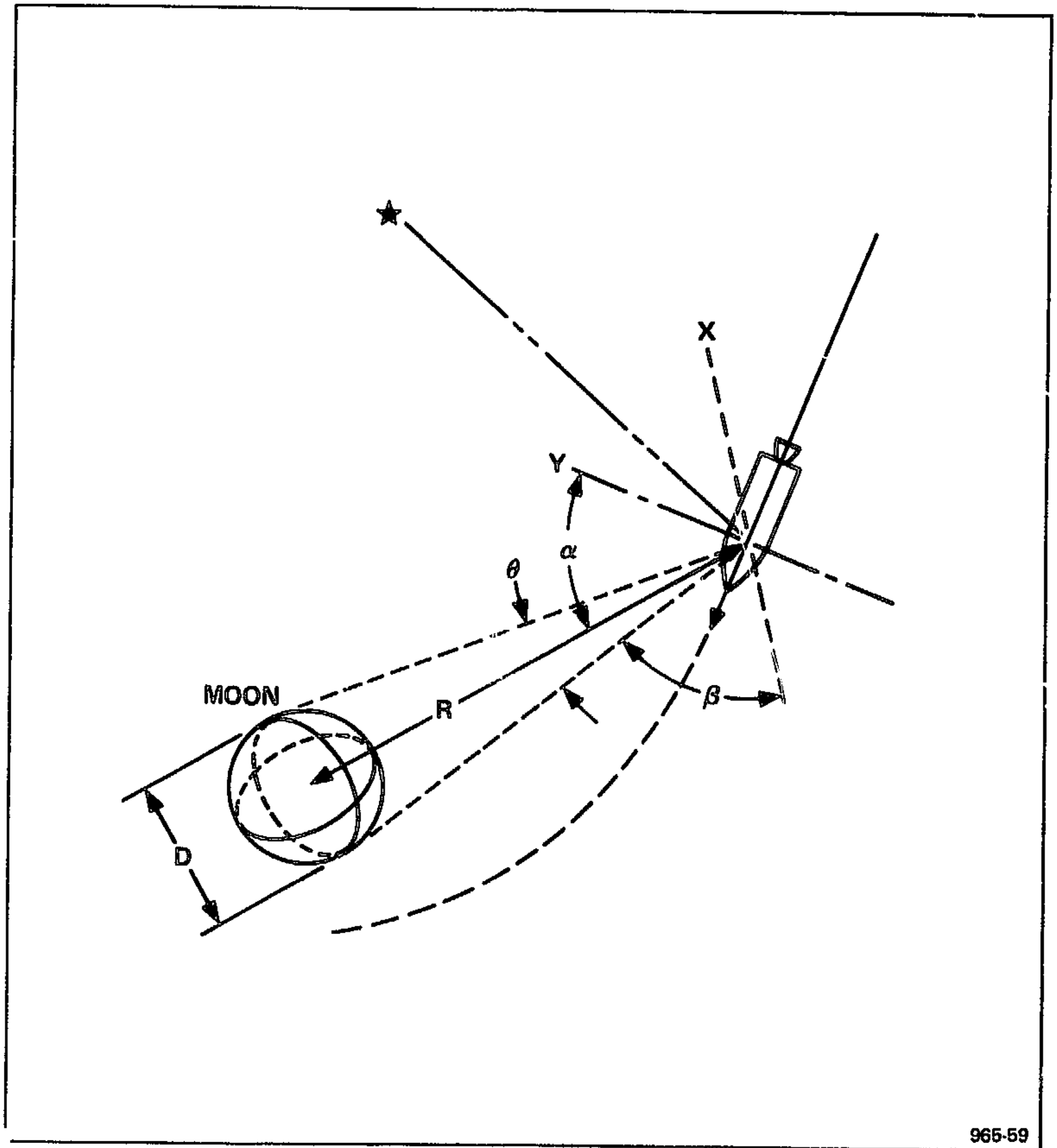

Figure 4-9. Lunar Approach Sensor Measurements

Given r, $\dot{r}$, and $\dot{\theta}$, the specific angular momentum is: $p = r^2\dot{\theta}$

and the specific energy is: $U = 0.5\ (\dot{r}^2 + p\ \dot{\theta}\ ) - \mu/r$

then the eccentricity is

$$e = [1 + \frac{2p^2U}{\mu^2}]^{1/2}$$

and the perilune radius is

$$r_p = \frac{p^2}{\mu(1+e)}$$

Using the above expressions, the sensitivity of perilune altitude to the measured value of $\dot{r}$ and $\dot{\theta}$ for a typical case is shown in Table 4-6.

These sensitivities show that even with quite crude measurements, injection can be made into a lunar orbit. If the error in measured quantities can be held to 2%, the lunar orbit can be close enough to permit acquisition of the lunar space station by radar (300 NM range and assuming optimum space station position), so that the operation is autonomous. If sensors are not available that can measure range and range rate within 2%, use of ground stations to obtain range and range rate data with the onboard sensor providing the angular line-of-sight data is another possible approach.

4.3.2 Plane Change

Because of orbital precession and various mission requirements, the space tug must be prepared to inject into any lunar orbit. Target orbits at TLI must contain the sub-earth point. By selecting a suitable target orbit the plane change required to attain any lunar orbit can be up to 90 degrees. Injection can be made without plane change at the moon if a wait of up to two weeks is acceptable.

Table 4-6. Perilune Sensitivity to Sensor Errors

	r METER	$\dot{r}$ METER/SEC	$\dot{\theta}$ RAD/SEC	e	r_p km	Δr_p km
NOMINAL	37.04E6	1000.	0.46E-5	1.505	3242.	--------
r CHANGE	+5%	1000.	0.46E-5	1.603	3792.	+550.
$\dot{r}$ CHANGE	37.04E6	+5%	0.46E-5	1.560	3172.	-70.
$\dot{\theta}$ CHANGE	37.04E6	1000.	+5%	1.549	3512.	+270

The impulse required to make the plane change decreases with increasing altitude. Hence, a general procedure is as follows: First inject into a high altitude lunar orbit, perhaps 2000 km. A plane change is next made into the plane of the final orbit. Next is a wait for proper phasing to occur, and finally execute a Hohmann transfer into the final orbit. If the navigation and guidance accuracy is limited, a second Hohmann transfer may be required to correct the error of the first.

The plane change operation can be done with a single burn. In the notation of Figure 4-2, wait until the tug reaches the intersection of the two orbit planes; this occurs when Z is zero. At this time an impulse is applied to force Z to zero.

Because ϕ_Z is independent of ϕ, there is no correlation of time for making the plane change with the time for making the Hohmann transfer; so, the plane change must be treated as a separate burn in general.

The properties of circular lunar orbits of interest for phasing are shown in Table 4-7. It is assumed that the lunar space station is in a 60 NM orbit. The synodic period is the maximum time required for phasing. A high altitude is advantageous both for reducing the phasing time and for reducing the impulse required to make a plane change.

The period of an orbit is given by

$$T = 2\pi \sqrt{\frac{a^3}{\mu}}$$

where a is the semi-axis major.

For the moon,

$$r = 1738.\ \text{km}$$

$$\mu = 4.9027779 \times 10^3\ \text{km}^3/\text{sec}^2$$

The synodic period, T, is given in terms of the periods of two orbits by

$$\frac{1}{T} = \frac{1}{T_1} - \frac{1}{T_2}$$

Table 4-7. Synodic Period For Circular Lunar Orbits

ORBITAL ALTITUDE (NM)	ORBITAL PERIOD (MINUTES)	SYNODIC PERIOD (HOURS)
60	118.920	—
100	126.137	34.64
300	164.279	7.17
500	205.640	4.69
1000	321.698	3.14
2000	600.406	2.47

4.4 RENDEZVOUS AND DOCK

The rendezvous phase is a terminal guidance phase and is to place the chaser vehicle in an optimum position and attitude for the docking phase. This optimum position consists of placing the chaser within visual range (for manned missions) and close enough to the target so that essentially a straight line trajectory may be used during docking. A series of vernier burns are normally used for this phase.

Docking requirements are dependent on the type and/or mechanism use for docking. The Apollo and Gemini docking requirements are shown in Table 4-8 for reference.

4.5 ONBOARD COMPUTATION OF TRAJECTORIES AND TRAJECTORY CORRECTIONS

The computation of a precision nominal translunar trajectory is a difficult task. Significant effects are produced by the sun's gravity and lunar vibration. Therefore, ephemeris data must be available. No closed form solutions exist, so the trajectories must be numerically integrated. Some search process must be used to meet the end conditions, which is equivalent to running perhaps 3 or 4 trajectories with given initial conditions. For these reasons, it is not considered a reasonable task to generate precision nominal translunar trajectories onboard.

Missions which permit radar contact with the target, e.g., earth orbital operations or ascent from lunar surface to LOSS, can probably be done without a precision trajectory with little additional impulse requirement. Some extra time would be required for correction of the preliminary orbit.

In-depth analysis and trade studies are needed to determine the extent onboard navigation can be efficiently utilized for the tug design missions.

Table 4-8. Docking Requirements

PARAMETER	APOLLO TOLERANCES	GEMINI TOLERANCES
LONGITUDINAL VELOCITY (FT/SEC)	0.1–1.0	±1.5
VERTICAL AND LATERAL VELOCITY (FT/SEC)	± 0.5	±0.5
VERTICAL AND LATERAL DISPLACEMENT (FT)	± 1.0	± 1.5
RELATIVE ANGULAR MISALIGNMENT (DEG)	± 10.0	± 10.0
RELATIVE ANGULAR VELOCITY (DEG/SEC)	± 1.0	± 0.75

5.0 SENSOR SELECTION RATIONALE

The candidate configurations considered and the sensor selection rationale used are presented in the following paragraphs. Detailed cost trades were not performed since firm cost estimates were not available. These cost trades would be performed in future study efforts.

5.1 IMU

The IMU is an indispensable unit for navigation. Therefore, the decision to be made was not whether to use an IMU but whether to use a strap-down or gimballed unit. The strap-down configuration was selected for the space tug based upon the following rationale.

5.1.1 Trade-Off Considerations

The strap-down IMU has advantages over the more conventional gimballed unit in the areas of weight, power, size, simplicity, reliability, reaction time, and its adaptability to either redundancy techniques or to updating with improved inertial sensors. The disadvantages of a strap-down IMU are its increased computational requirements, lower accuracy and calibration difficulties when mounted in the user vehicle.

A comparison of the strap-down hexad with a triad strap-down, a triad gimballed system, and four commercially available carousel IV IMU's is shown in Table 5-1. The comparison demonstrates the advantages of the strap-down mechanization on the basis of size, weight, power, system cost, and MTBF for comparable performance. Of particular significance is the reliability advantage which arises both from the fact that the strap-down IMU has no moving parts other than inertial components and from efficient redundancy mechanization provided by the hexad configuration. The reliability for various IMU configurations is plotted versus the instrument loop reliability time constants (λt) in Figure 5-1 (Reference D-8).

Table 5-1. Strapdown Versus Gimballed IMU Trade Matrix

CANDIDATE	REDUNDANCY	SIZE $(FT)^3$	WEIGHT (LBS)	POWER (WATTS)	COST ($K)	MTBF* (HRS)	NAVIGATION ERROR (NM/HR)
STRAPDOWN HEXAD	SINGLE IMU 6 GYROS 6 ACCEL.	1.6	80	200	200	4000	1.5
CAROUSEL IV (BOEING 747)	4 UNITS 12 GYROS 12 ACCEL.	5.0	212	1380	480	2500	2.0
STRAPDOWN TRIAD	SINGLE IMU 3 GYROS 3 ACCEL.	0.5	44	130	150	2000	2.0
GIMBALLED TRIAD	SINGLE IMU 3 GYROS 3 ACCEL.	1.3	74	345	120	1500	1.5

* MTBF FOR SINGLE IMU

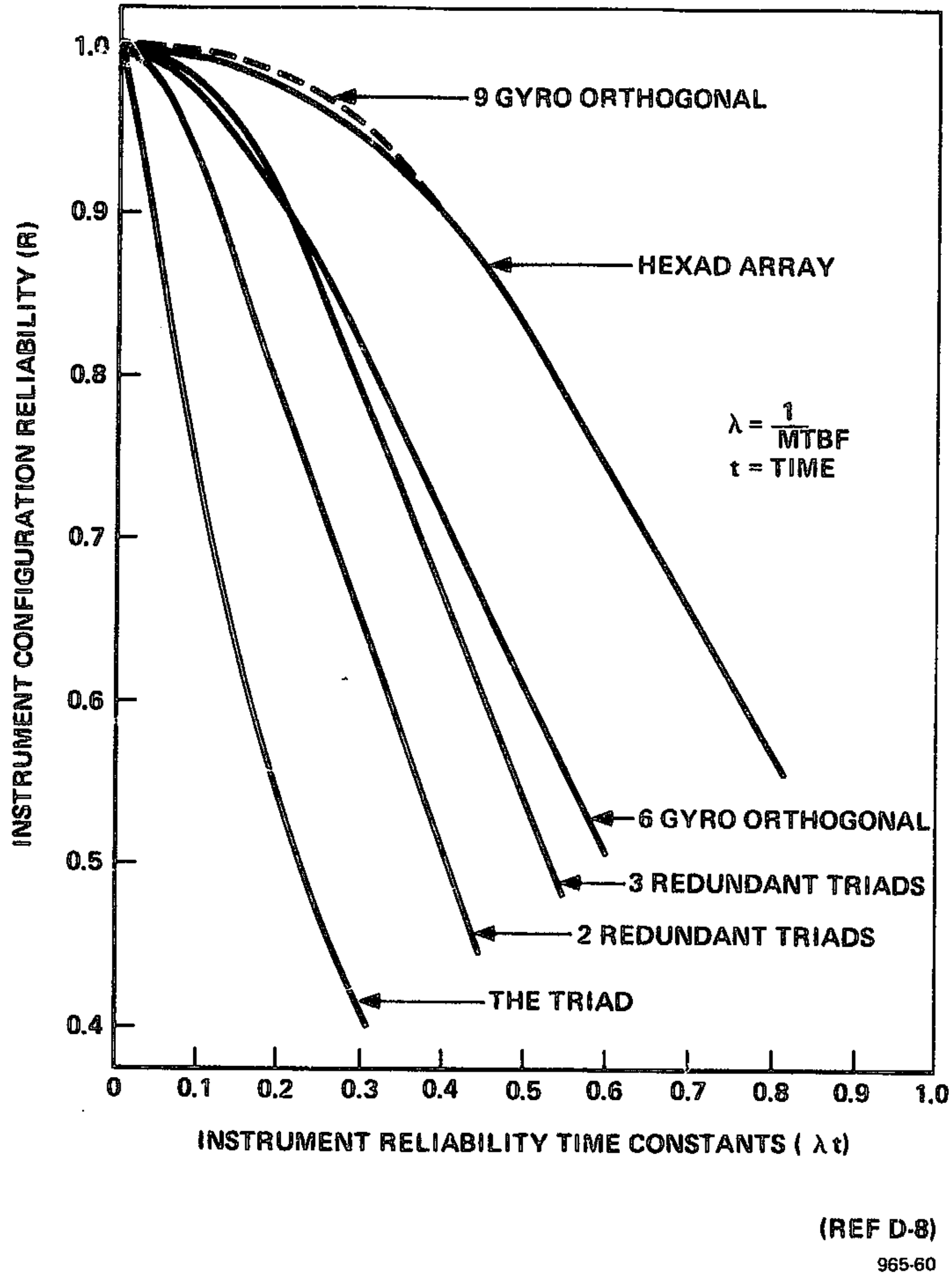

Figure 5-1. Reliable Plots – Perfect Failure Isolation

A significant advantage of the strap-down configuration is the ability to add redundancy by simply adding inertial sensors. Analysis has shown (Reference D-1) that six gyros and six accelerometers, whose input axes are colinear with the normals to the face of a regular dodecahedron, provides a backup capability that exceeds that of triple redundant orthogonal configurations. This mechanization is achieved with only a modest increase in weight and power over the conventional three orthogonal sensor package as shown below.

	WEIGHT (LBS)	POWER (WATTS)
3 sensor configuration	44	130
6 sensor configuration	80	200

Another advantage is that a strap-down IMU has complete rotational freedom in all axes, while the gimballed platform is limited in at least one axis, usually Yaw.

For space tug missions which operate only in the space domain, the disadvantages of the strap-down are minimized. For example, there is no requirement for the IMU to operate during atmospheric flight. Thus, the vibrations and buffeting associated with ascent or descent through the atmosphere is not a problem. However, for the four stage Saturn V mission, which include an ascent through the earths' atmosphere, vibration and vehicle rates will result in degradation of strap-down performance. Although the performance of the strap-down is degraded, a strap-down IMU's accuracy is sufficient to insert into a 100 nmi waiting orbit. For those missions that require additional impulses to achieve new orbits or trajectories, a navigation update and IMU alignment might be required.

Another disadvantage of the strap-down is the inability to calibrate the system in the launch configuration. This requires that the inertial sensors have long-term stability of error coefficients. Intermittent recalibration of the inertial sensor error coefficients could bound this problem.

In space, only gyro and accelerometer bias values can be determined and the calibration techniques are equally applicable to the strap-down and gimballed IMU.

The computation rates required for updating the direction cosines are a function of the maximum vehicle turning rates and vehicle coning which are not yet available. It is expected that a 32-bit second order algorithm operating at 100 updates per second will provide the required accuracy. This is well within the capability of present-day general-purpose computers.

5.1.2 System Description

The IMU description presented below is for a redundant sensor Strap-Down Inertial Reference Unit (SIRU) configuration developed by the MIT Instrumentation Laboratory as presented in Reference D-8.

The SIRU Inertial Subsystem is configured as a multiple-sensor non-orthogonal Inertial Component Sensor Assembly (ICSA) that is complete with redundant supporting electronics. The inertial-component package employs a redundant implementation using six single-degree-of-freedom gyroscopes and six linear accelerometers. Each is operated in a classical pulse-torque-restrained strap-down control mode.

Within the inertial-component package, instrument input axes are arranged in a unique pattern that corresponds to the array of normals to the faces of a regular dodecahedron, as shown in Figure 5-2. This array was selected because it is completely symmetrical; i.e., the acute angle (2 a) measured between any 2 axes is equal (63.4°) for all axes combinations. This symmetry allows realization of maximum redundancy with optimal performance at each level of redundant operation. It provides the basis for the high-reliability SIRU formulation.

Since all input axes are non-orthogonal, each instrument's output contains a measure of redundant data. This data is processed to obtain a measurement solution that flexibly adapts to account for failures of instruments or corresponding electronics. The unique symmetry of the measurement array allows signal-flow processing to provide satisfactory system performance with any three gyro or accelerometer failures. Further, implementation of self-contained failure detection and isolation allows isolation to any two gyro or accelerometer axes and detection of a third failure.

The system has been formulated in a strap-down configuration to take maximum advantage of the relative ease of adaptation of the failure-isolation equations and the data processing to a digital computer, as well as the inherent advantages of improved reliability, accessibility, size, and weight. The processing and storage overhead associated with the triad-solution form and failure detection are negligible.

The ICSA electronic mechanization has been redundantly configured to provide supporting instrument electronic functions that are free from single-point failure mechanisms. Figure 5-3 illustrates the basics of the mechanization. Functional axes have been defined, corresponding to each measurement axis, which consist of a gyro and accelerometer module supported by common ac and dc power supplies. These supplies are repeated on a per-axis basis so that a singular loss does not hazard the redundant measurement capabilities. Also, each gyro and accelerometer module includes a temperature controller and torque-to-balance control loop which further extends the axis redundancy.

The electronic-system configuration has been mechanized to provide the necessary failure-isolation characteristics with comparative redundant application based on the relative reliability of the particular functional elements. For example, the estimated failure rate of a DC power supply is less than 10 per million hours, while a gyro module estimate might range between 100 and 200 per million hours, dependent upon the instrument and torque electronic configuration. Thus, for the DC power supply, dual redundancy is sufficient and does not compromise the end-to-end system reliability index.

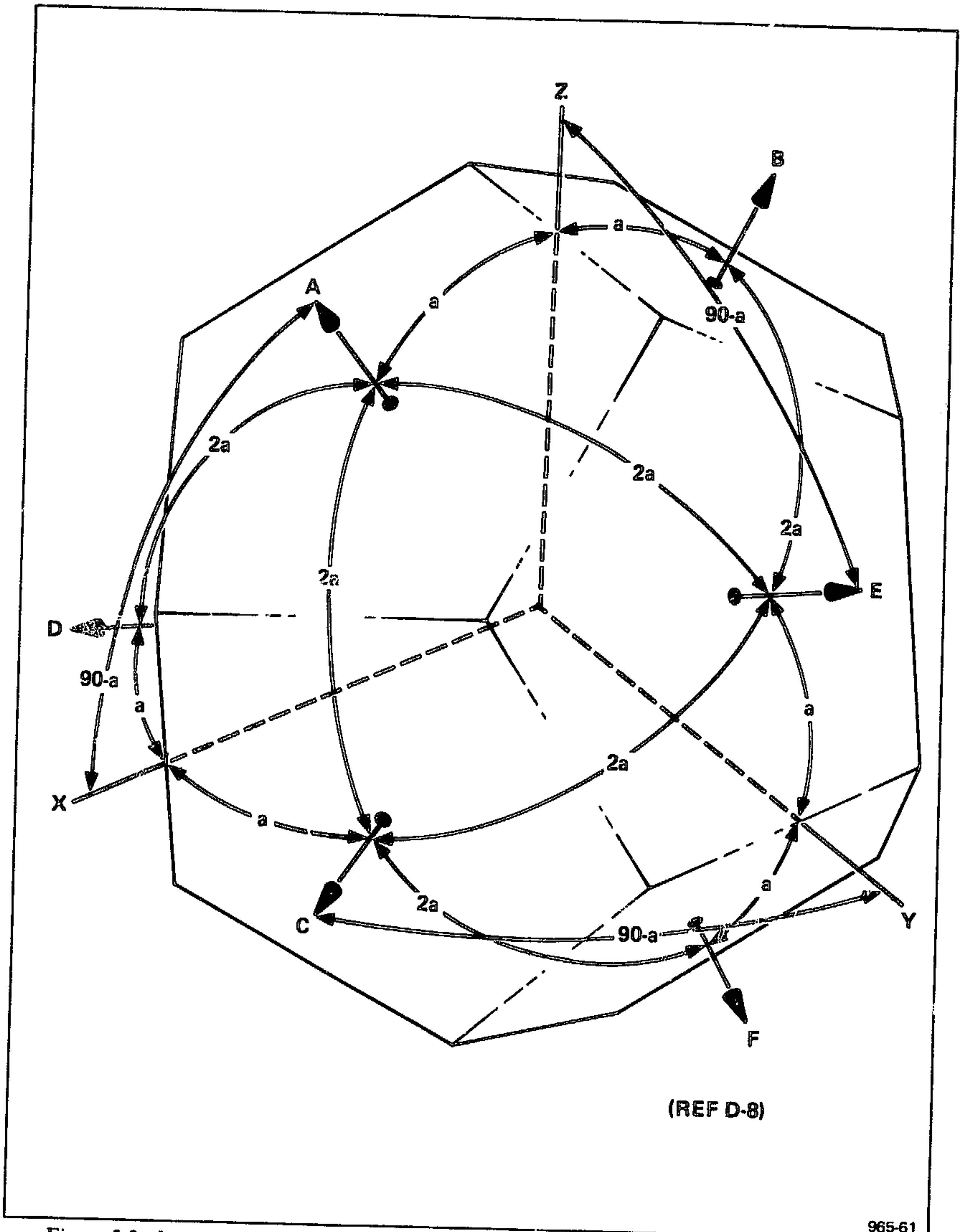

Figure 5-2. Instrument Input Axes Orientation Relative to the Instrument Frame Triad and the Dodecahedron

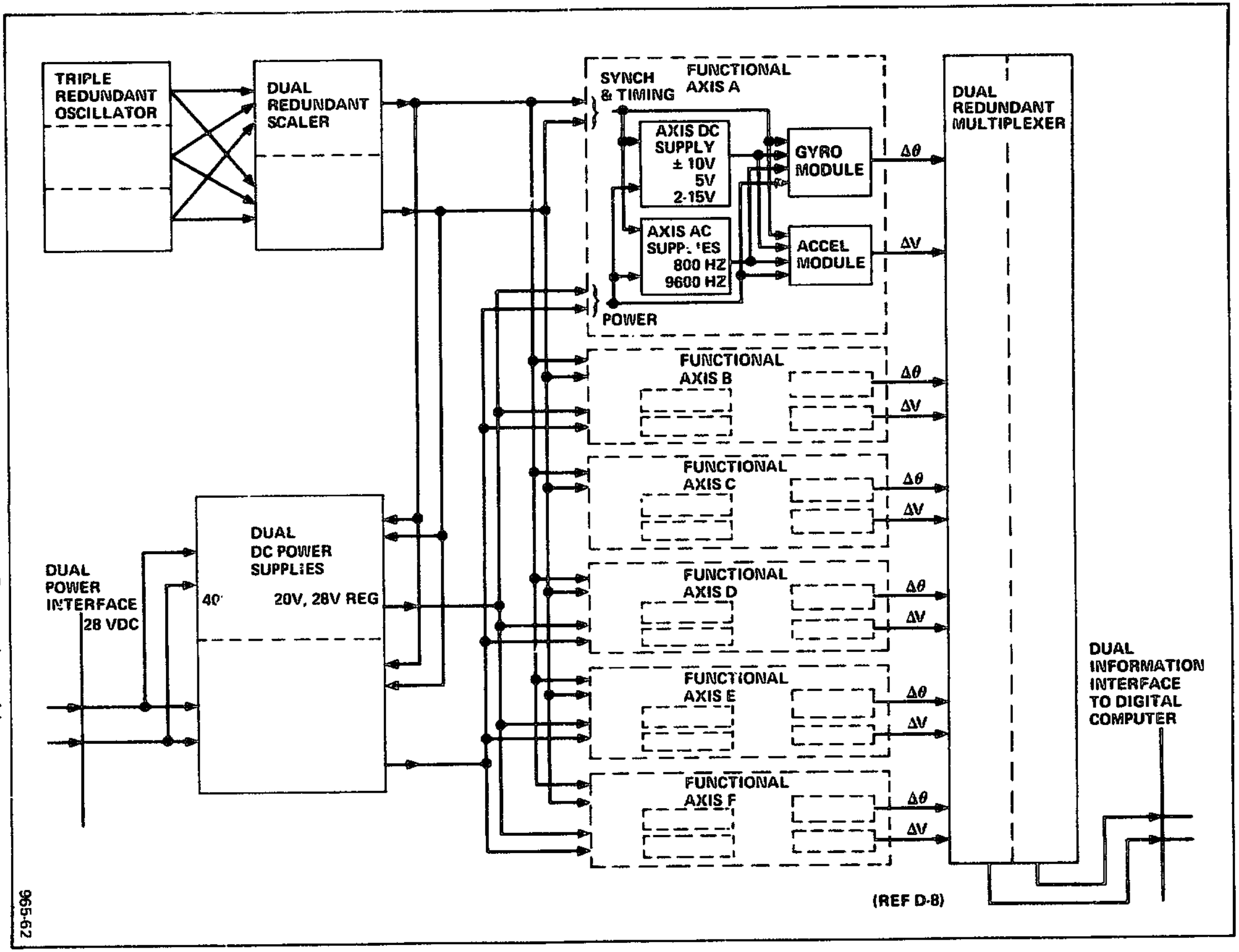

Figure 5-3. Block Diagram of Siru Inertial Component Sensor Assembly

5.2 RENDEZVOUS RADAR

In selecting a rendezvous radar, the choice was between an electromagnetic unit of the type presently used on LEM and a Laser Radar developed by ITT under direction of NASA MSFC. The Laser Radar System was selected. The factors that contributed to this conclusion are discussed below.

5.2.1 System Trade-Offs

Optical (laser) radars have three major advantages over the microwave radar. These are:

1. The wave length (λ) of microwave systems is typically 1×10^{-2} meters as compared to approximately 1×10^{-6} meters for optical systems. Thus for the same beam width, considerably smaller antenna size is needed for the optical laser system.

2. The receiver side lobe pattern in optical systems can typically be reduced to 60-70 db below the main lobe. This is compared to 20-30 db for the microwave system. The narrow beam width of the laser radar results in lower power requirements than the microwave system. Also, the side lobes can produce erroneous information as a result of their reflection off the lunar surface or target vehicle.

3. With the optical radar, the transmitter and receiver can be scanned over a field-of-view much larger than its own beam width in a relatively simple fashion. The present laser system has the capability to scan the transmitter and receiver over a 30 degree by 30 degree field-of-view by using piezoelectric beam steerers and amplifying optics. Except for phased arrays, which are relatively large and power consuming, the microwave scanning radar systems must depend on mechanical gimbals for scan coverage.

One of the prime factors in selecting the laser radar is its applicability to automatic docking as well as rendezvous. For automatic docking, the sensor must be capable of accurately measuring range, range rate, and relative vehicle attitude right up to vehicle contact. The laser system will perform accurately to approximately zero range while the microwave system is not accurate at a range of approximately 50 feet or less. The estimated system performance characteristics for the laser radar system are given in Table 5-2. Table 5-3 compares some of the more significant characteristics of the laser and microwave radar systems.

Table 5-3 points out some additional advantages and one disadvantage of the laser system. The weight of the laser system, including chaser and target equipment, is 50 lbs as compared to 85 lbs for the microwave systems.

One disadvantage of the laser radar as presently designed is the maximum range of 75 NM as compared to 405 NM for the electromagnetic unit. Although rendezvous can be performed with acquisition at this range, it may be desirable to have longer range capability. Increased acquisition range requirements result from transfer orbit injection accuracy uncertainties. Rendezvous fuel requirements as a function of injection dispersions can result in significant fuel penalties when the acquisition range is limited. A longer range can be provided with a more efficient detector which has been demonstrated at a component level in the lab and can be incorporated into the system by the '72 to '73 time period.

Table 5-2. Estimated System Performance Characteristics For Scanning Laser Radar System*

RANGE	0–120 KM (75 MILES)
RANGE ACCURACY	± 0.02% OR ± 10 CM (WHICHEVER IS GREATER)
RANGE–RATE	0-5 KM/SEC. (11,200 MPH)
RANGE–RATE ACCURACY	± 1.0% OR ± 0.5 CM/SEC. (WHICHEVER IS GREATER)
ANGLE COVERAGE WITHOUT GIMBALS	± 15 DEGREE PITCH ± 15 DEGREE YAW ± 90 DEGREE RELATIVE ROLL
ANGLE ACCURACY	
PITCH AND YAW	± 0.02 DEGREE
ROLL INDEX	± 1.0 DEGREE
ACQUISITION SCAN TIME	1 TO 150 SECONDS
ANGLE–RATE	
ACQUISITION MODE	0–0.4 DEGREE/SECOND
TRACK MODE	0-10 DEGREE/SECOND
ANGLE–RATE ACCURACY	± 1.0% OR + 0.01 DEGREE/SECOND (WHICHEVER IS GREATER)

* ASSUMES A COOPERATIVE TARGET

Table 5-3. Laser VS Microwave Rendezvous Radar

	OPTICAL-LASER	MICROWAVE
RANGE ACCURACY (1 N. M. RANGE)	± 1.2 FT	± 80 FT
RANGE (MAX.)	75 N. M.	405 N. M.
RANGE (MIN.)	0 FT	50 FT
SIDE-LOBES	-60 to 70 DB	20 TO 30 DB
POINTING	PIEZO-ELECT DEFLECTOR	ANTENNA GIMBAL
SYSTEM WEIGHT (CHASER AND TARGET)	CHASER 28 LBS TARGET 22 LBS	85 LBS
SYSTEM POWER (CHASER AND TARGET)	CHASER 30 WATTS TARGET 15 WATTS	235 WATTS

* ASSUMES COOPERATIVE TARGET

No data is presently available on the reliability and service life of the laser unit.

5.2.2 System Description

The system discussed herein is a lightweight Scanning Laser Radar (SLR) system that is being developed for NASA by ITT (References D-5 and D-6) to generally address the problem of rendezvous, docking and station keeping. The SLR can also be used for other mission operations such as a lunar landing aid. The SLR does require that the target be cooperative; a four-inch optical corner cube reflector mounted on the target will allow the chaser to cooperatively acquire and track the target from a maximum range of approximately 120 km (75 miles). The maximum range of the system is limited to approximately one mile against a non-cooperative diffuse target (i.e., the outer skin of a space vehicle). The return signal off the cooperative corner cube reflector is several orders of magnitude stronger than a non-cooperative diffuse return off any portion of a vehicle target, therefore, it is relatively easy to acquire and maintain track on the designated corner cube reflector. In addition, the laser radar using the corner cube reflector can operate even with a sunlit cloud background.

The target equipment consists of the corner cube reflector and an optical receiver that measures the angular direction of the incoming laser beam, while the other vehicle equipment consists of a laser transmitter-receiver for range and angle measurements. Both vehicle equipments are accurately positioned and oriented relative to the vehicle axes. Thus, the SLR measurements can be converted to vehicle coordinates as required for rendezvous and docking.

The SLR is made up almost entirely of digitally controlled devices and digital electronics, including the two scanners that position the transmitter and receiver. The radar scan patterns for both the acquisition and track modes are digitally stopped. This feature allows for pre-programming of the scan patterns and/or for the remote control of the scan by a computer located onboard the space vehicle.

The radar transmitter-receiver sensor package weighs approximately 20 pounds. The radar transmitter used in the SLR is a semi-conductor laser that is about the size of a cigarette package. The beam steerer is made up of solid state devices and optical lenses. The radar receiver is made up of optical lenses and filters and an image disector.

Functional block diagrams of the chaser and target equipment are shown on Figures 5-4 and 5-5, respectively.

The laser radar system, as presently designed, will scan a field-of-view 30 degrees by 30 degrees without the use of gimbals. However, in order to employ the same system for rendezvous and as a landing aid, gimbals will be required since the target directions will differ by more than 30 degrees with respect to the space tug axes. A simple indexed gimbal system can be employed which can be latched at discrete angles. This will provide the attitude flexibility required while preserving the scan feature. The added system weight is expected to be no more than 6 to 8 pounds. A disadvantage of the laser radar when used as a landing aid is that it cannot penetrate dust blown up by the descent engines. Therefore, navigation close to the lunar surface probably would be accomplished using only lunar landing radar inputs. However, if corner reflectors are located at the periphery of the landing site and at a sufficient height above the lunar surface, a laser radar could navigate the landing vehicle to touchdown.

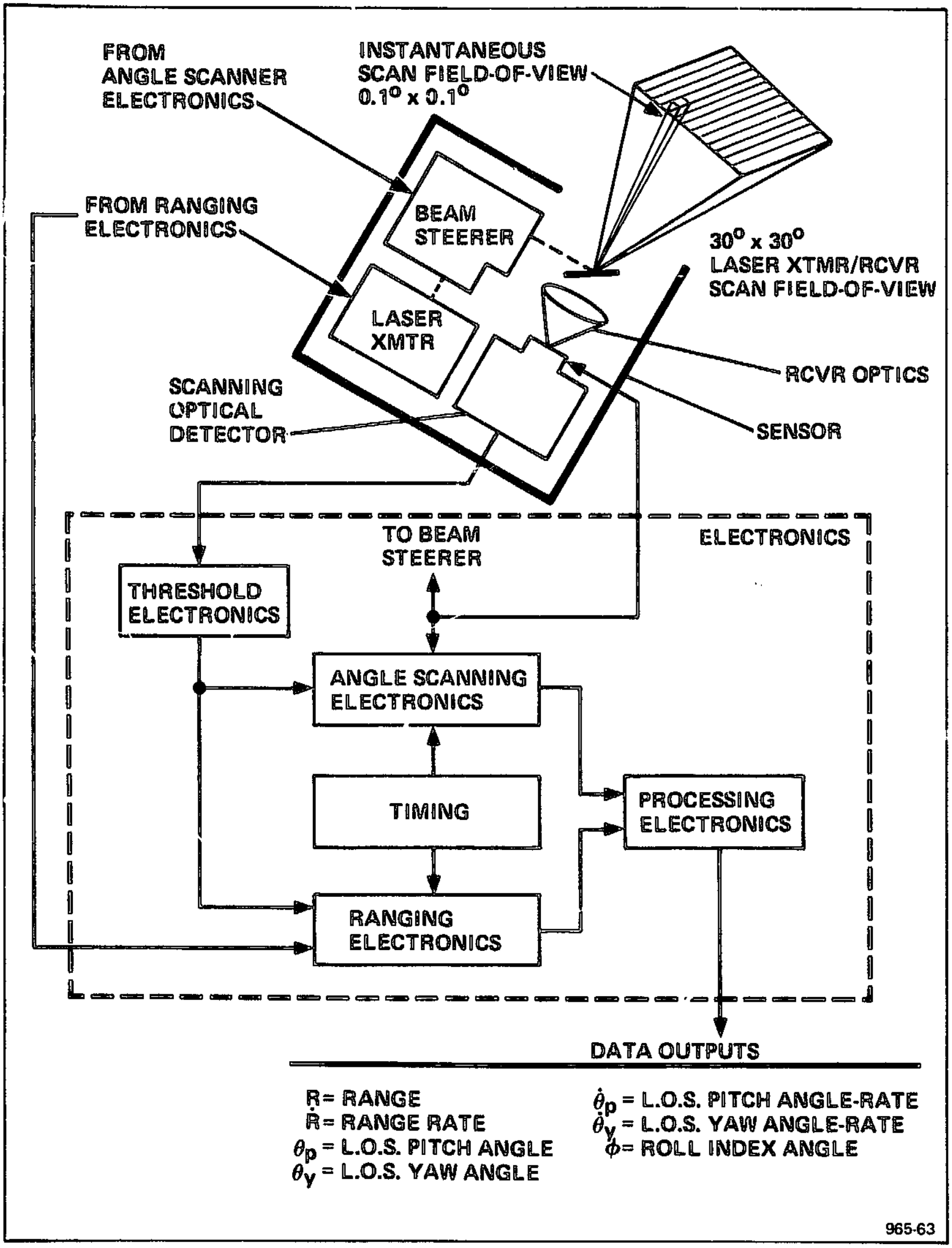

Figure 5-4. Scanning Laser Radar Block Diagram

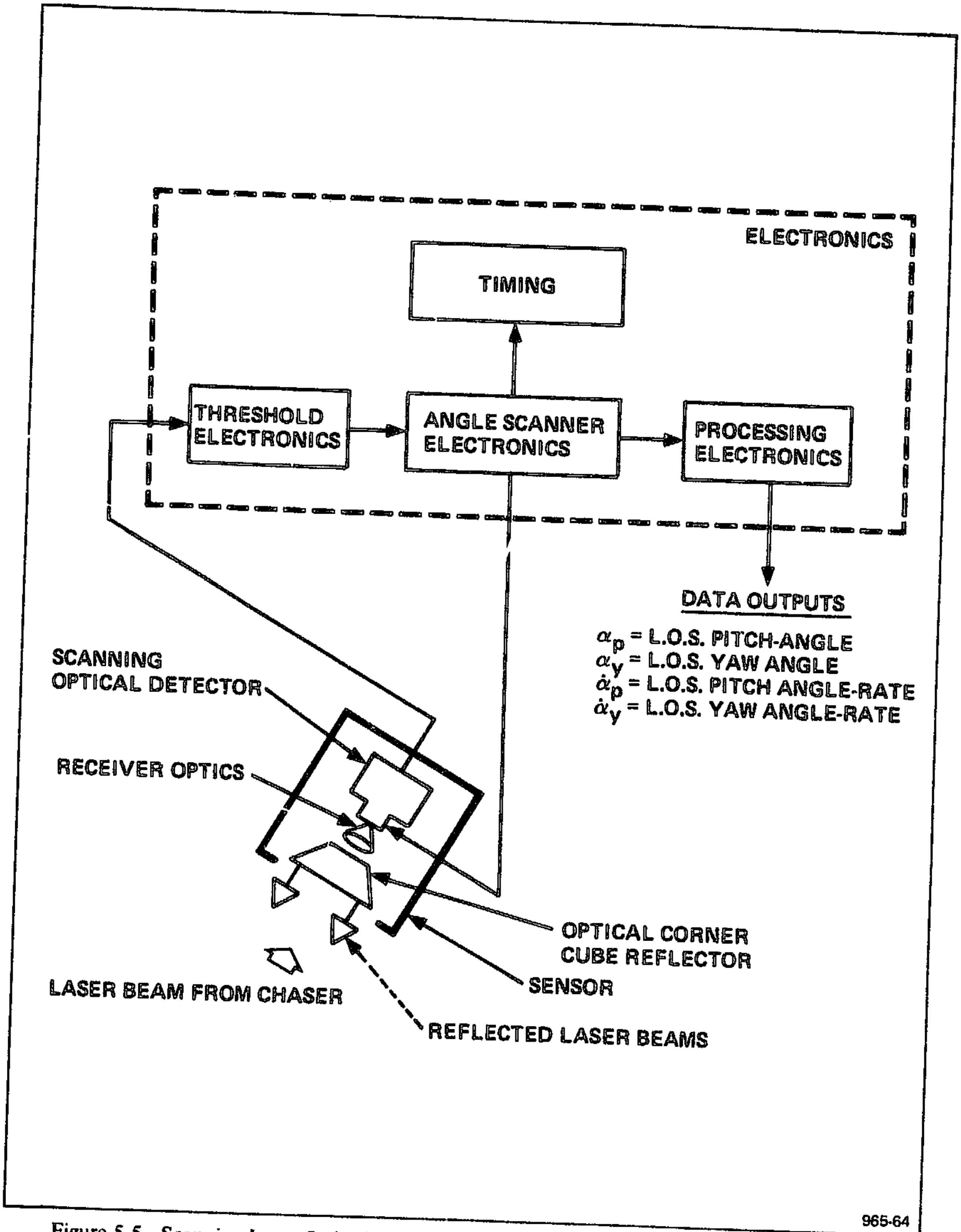

Figure 5-5. Scanning Laser Radar Block Diagram

5.3 STAR TRACKER, LANDMARK TRACKER AND HORIZON SENSOR

Two star trackers, a landmark tracker, and horizon sensor were selected for autonomous navigation and IMU alignment. The two star trackers are used for IMU alignment, while the star trackers coupled with the landmark tracker or horizon sensor are used for navigation update.

It should be noted that two star trackers are not mandatory. Both alignment and navigation updates can be performed with one star tracker by sequentially acquiring and taking readings on two or more stars. The advantage of two units operating simultaneously is that less time is involved and greater accuracy is achieved because all readings are taken from the same vehicle state. Also, in case of failure of one unit, the second unit can be operated as stated above, thus providing system redundancy.

5.3.1 Configurations Considered

The combinations considered for IMU alignment were:

1. Horizon scanners and one star tracker.
2. Horizon scanners and sun sensor.
3. Horizon scanners and gyrocompassing.
4. Two star trackers (selected subsystem).

The subsystem combinations considered for navigation update were:

1. Horizon scanner with two star trackers.
2. Horizon scanner and landmark tracker.
3. Two star trackers and landmark tracker (selected subsystem).

5.3.1.1 Configurations for IMU Alignment

The horizon scanner – star tracker combination for in-orbit IMU alignment will provide a vertical alignment, with respect to local vertical, of approximately 7 arc minutes and azimuth alignment, with respect to an inertial coordinate frame, of approximately 40 arc seconds. This mode of alignment is satisfactory for earth and lunar orbit, but is insufficient for alignment during translunar or transearth coast or while on the lunar surface. However, alignment on the lunar surface can be performed using the IMU accelerometers for vertical alignment and the star tracker for alignment in azimuth as is presently done with the LEM.

The second combination: horizon scanners and sun sensors, is essentially the same as the first. It is limited, however, because of the single object available for reference.

The third alignment technique employs horizon scanners for local vertical alignment and gyrocompassing for azimuth alignment to the orbit plane. This technique will provide in-orbit alignment accuracy to approximately 8 arc minutes in each axis. This technique is not applicable for azimuth alignment on the lunar surface or for translunar or transearth coast because of the low lunar gravity potential.

The fourth and selected subsystem is two star trackers which will align the IMU to an inertial coordinate frame. For alignment, the star trackers acquire, lock on, and track known stars which are at least 30 degrees apart. This technique, which parallels the method used for the CSM and LM, is accurate to approximately 40 arc seconds or less, and has the advantage of quick reaction. Furthermore, the two star trackers operating with the landmark tracker provide an excellent method for acquiring navigation updates as discussed below.

5.3.1.2 Configurations for Navigation Update

Figures 5-6 and 5-7 show the velocity and position accuracies achievable for 5 different subsystem combinations including the 3 considered. The errors shown are for a vehicle in a 100 NM earth orbit. This same accuracy or better can be achieved in a 60 NM lunar orbit. However, the accuracy will be less for orbits above 100 NM.

The relative navigational accuracies achievable for the different subsystem combinations are dependent upon the orientation source (stellar or horizon) and the object tracked (landmark or local vertical). Landmark tracking can be performed more accurately than tracking the local vertical (horizon scanning). Thus, subsystem combination 3 is more accurate than subsystem combination 1 as shown on Figures 5-6 and 5-7. Also, a stellar orientation frame is superior to a local vertical frame because of the better pointing accuracy of a star tracker as compared to a horizon sensor. Therefore, subsystem combination 3 is superior to subsystem combination 2.

The star tracker-landmark tracker combination was selected because of its relative fast response and accuracy plus the fact that the same star tracker subsystem can be employed for alignment. As shown on the figures, the accuracy and response of this system approaches that for MSFN tracking.

This configuration will provide the accuracy required for low earth and lunar orbit missions and the lunar landing mission. The reliability and accuracy of navigation updates using the proposed subsystem beyond 100 nm and less than 100 nm altitude is dependent on the following conditions:

- Accuracy is limited by the knowledge of the locations of landmarks being used.
- The availability of landmarks is dependent upon:
  - Size, shape, and color contrast.
  - Local weather conditions.
  - Slant angle to target which subtends a smaller portion of the field of view.
  - Optical resolution of targets reduces number of targets as attitude above planet increases.

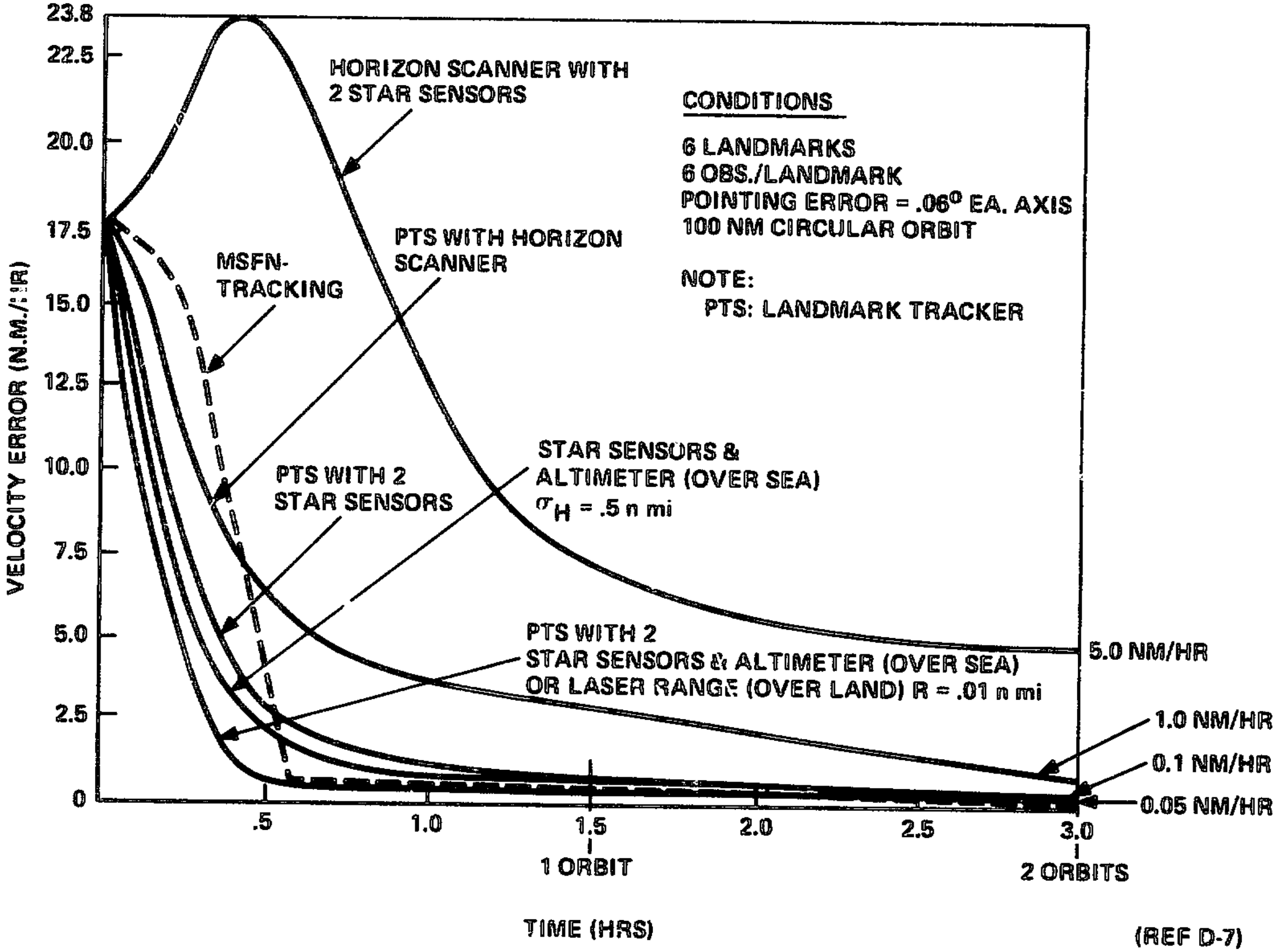

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Figure 5-6. Velocity Error For Five Different Systems

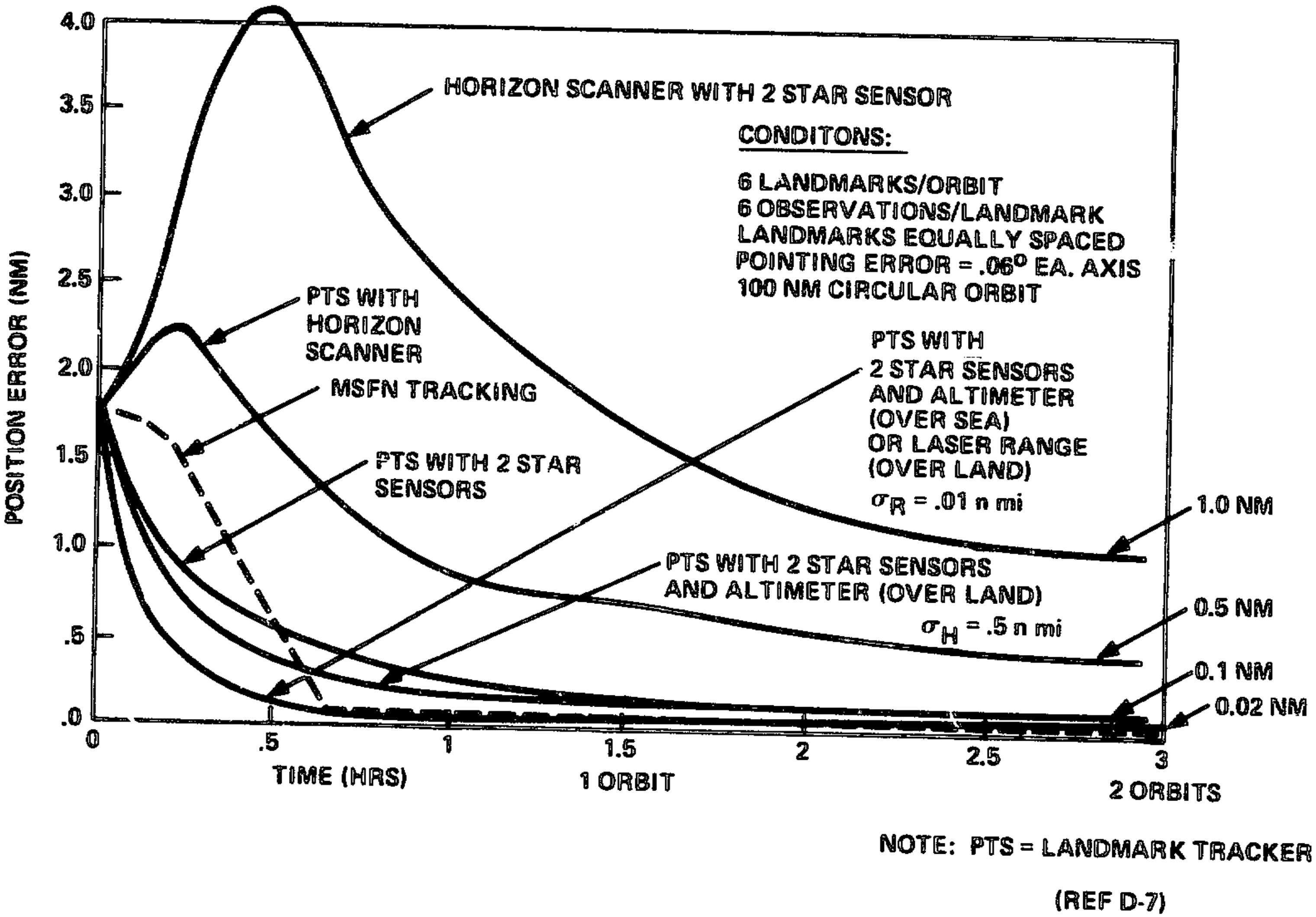


Figure 5-7. Position Error For Five Different Systems

For these reasons, the Apollo navigation during translunar-transearth flight used the visible horizon as a reference instead of landmarks.

Beyond 20,000 kilometers (10,000 nm) the navigation optical instrument errors begin to predominate. Also the number of available landmarks has reduced significantly. The uncertainty of cloud error over the remaining landmarks is increasing. These conditions dictate the necessity for a more reliable means of navigation update beyond this attitude. A horizon sensor/star tracker combination although less accurate than the star/landmark subsystem is never the less uneffected by cloud cover because the horizon sensor views the observable infrared horizon. Therefore, the automatic unmanned Space Tug missions whose orbits or trajectories exceed 10,000 nm in altitude will require a horizon sensor to perform navigation updates.

An added feature of the horizon sensor is that it also provides vehicle attitude information with respect to the local vertical which might be used as added inputs to the vehicle attitude control system.

5.3.2 Star Tracker Subsystem Description

Specific vendor units have not been selected. However, there are several designs available including some with solid state detectors. These devices have advantages over the more conventional photomultiplier sensors as follows:

- Less susceptible to background light.
- More reliable.
- Smaller.
- Requires less power.
- Makes electronic scanning feasible.
- Costs less.

The projection of star tracker development during the 1970's is for completely solid state units with no moving parts.

Star trackers typically consist of two units: the sensor subassembly and the electronic subassembly. The sensor unit which detects the light source (star) must be mounted such that it has optical access to the stars. (See Figure 5-8.)

A typical functional block diagram is shown in Figure 5-9. The direction of pointing is controlled by the computer via pointing commands to the sensor assembly. The initial pointing commands are determined from star position information stored in the computer. The sensor then hunts about this orientation until the selected star is detected. The light from the star is focused by the optics on the detector which in turn transmits an error signal to the signal processing electronics. There, the signal is converted to digital form and sent to the computer. The computer uses this signal to update the pointing commands in order to zero the error.

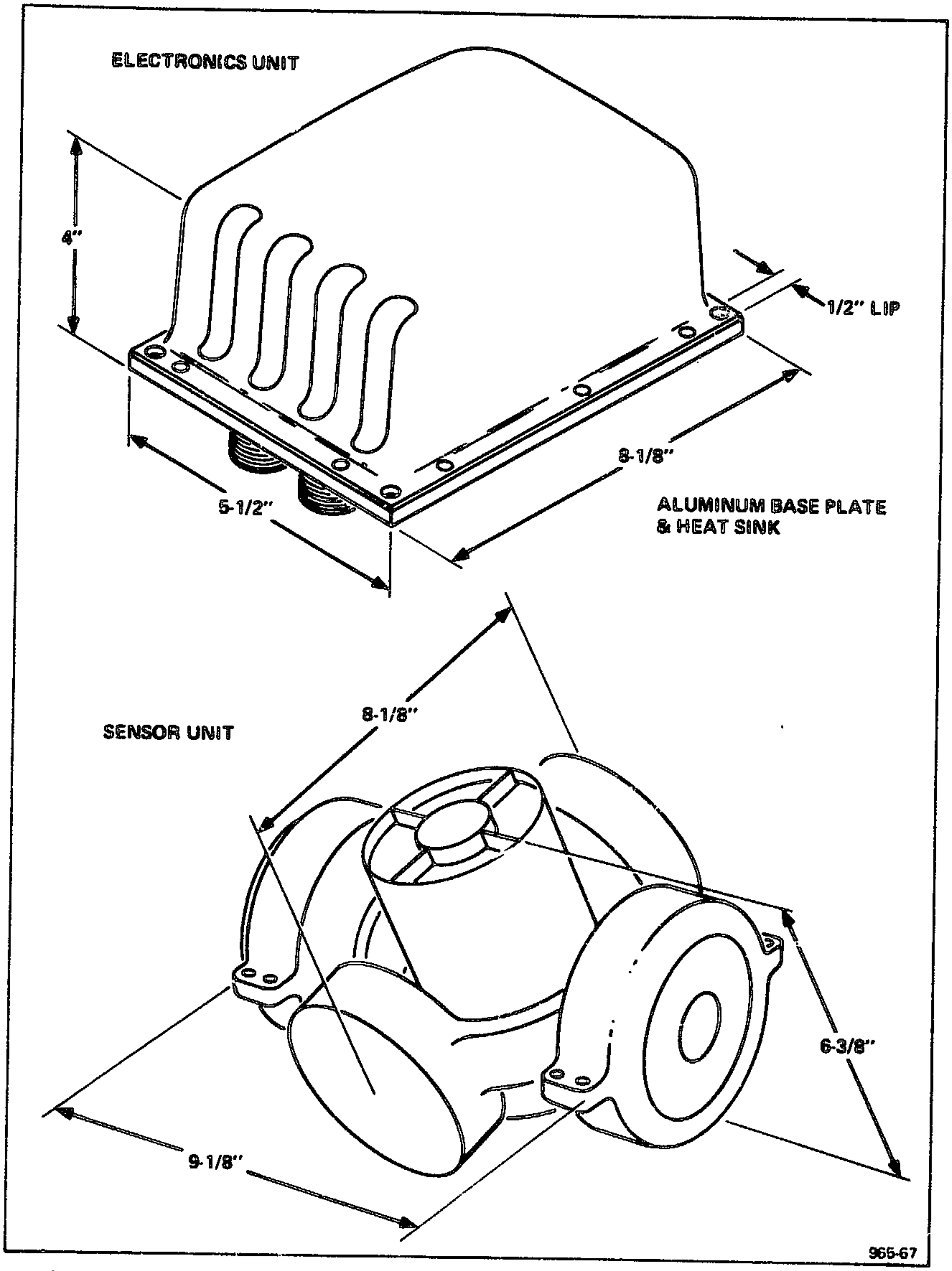


Figure 5-8. Representative State-of-the-Art Star Tracker (Litton Model - FM)

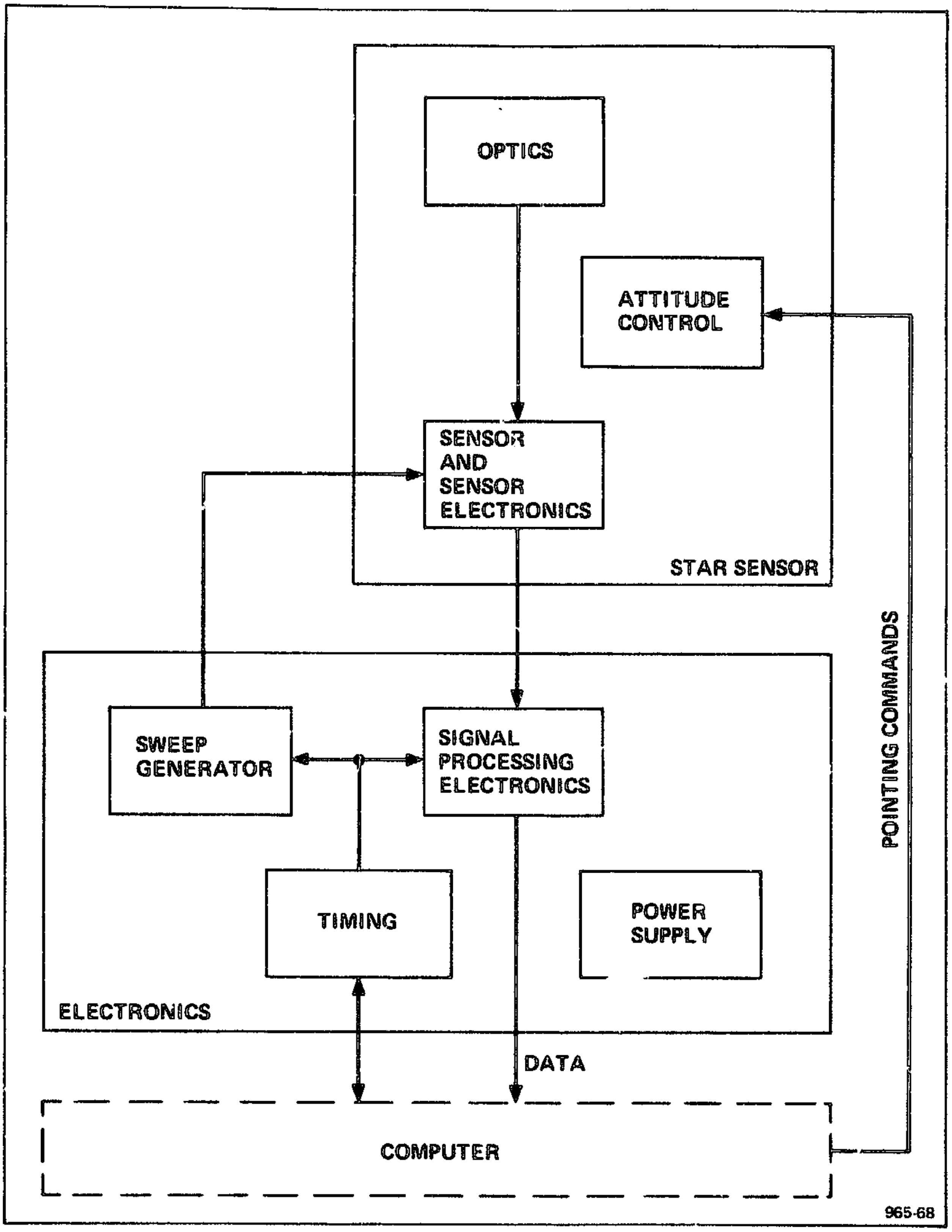


Figure 5-9. Star Tracker Functional Block Diagram

5.3.3 Automatic Landmark Tracker

The automatic landmark tracker or automatic earth feature sensor (AEFS) is similar to the star tracker in that it consists of a sensor unit and an electronic unit (See Figure 5-10). The entire sensor unit consisting of a vidicon and optics is gimballed in pitch and roll. Gimballing and optical magnification is required to obtain the desired accuracy. Gimballing is required for initial image motion compensation for acquisition and then final tracking: optical magnification of 2 or 3 power is required to overcome the TV line limitation for resolution. The gimbal gle readouts requirements are roughly 15 arc seconds requiring a simple inductosyn resolver and a 17 bit encoder. Table 5-4 lists the estimated performance characteristics.

5.3.3.1 Sequence of Operation

- Known Target Mode

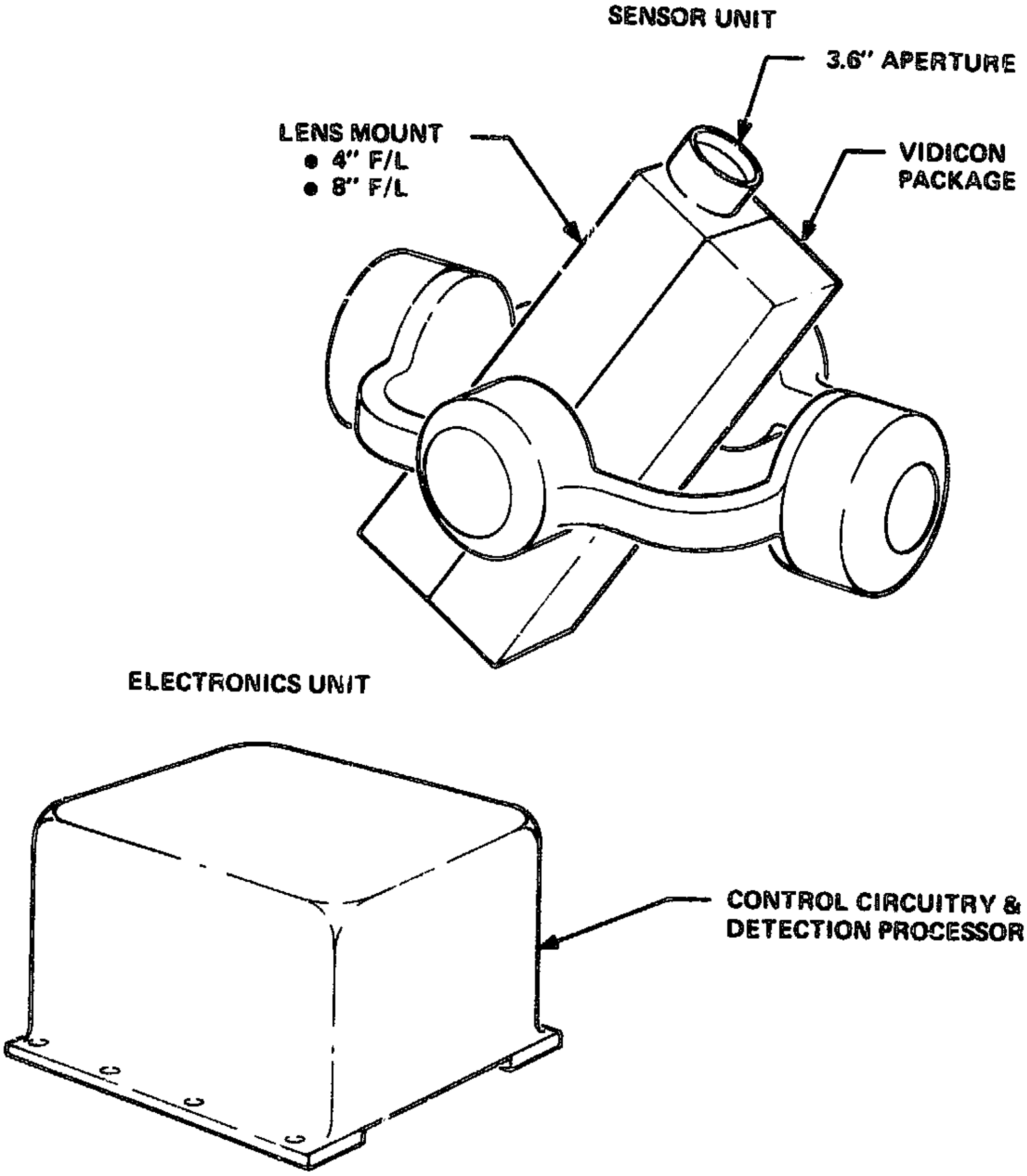
  The field-of-view is directed to the estimated location and tracked with estimated image motion rates. The shutter is opened, exposure made, and the video readout examined to set the circuits and logic levels. A second exposure is made and the video readout is processed to compute the x value. A third exposure is made and read out with the scanning lines orthogonal to the previous readout. The video is processed to compute y. These values define the roll and pitch pointing errors. The gimbals are corrected by the distance of x + y from boresight. A second sight is taken (two exposures) and a new x, y, value obtained. A succession of sights, taking two to four seconds per cycle will allow both pointing accuracy to one resolution element, and image motion compensation to one resolution element per cycle time.

  The only information which must be stored to identify a known landmark is the geodetic position. Target separation, size, contrast and other pertinent characteristics are accounted for in the selection process.

  During all coasting phase navigation, an extrapolation of position and velocity by numerical integration of the equations of motion is required. The integration scheme implemented dictates the integration increment required. Therefore, the number and location of landmarks must be chosen to maintain the required integration accuracy.

- Unknown Target Mode

  The unknown mode of operation is the same as the known mode, except that the sensor is pointed maximum forward along track and the estimated IMC applied if necessary. The video is examined to determine if a trackable target is within the field-of-view. When a trackable target appears, the operation is identical to the known mode.



WEIGHT: 30 LBS. (10-VIDICON, 12-GIMBAL, 8-ELECTRONICS)
VOLUME: 4 X 4 X 12 - VIDICON
4 X 4 X 12 - LENS MOUNT SINGLE UNIT
8 X 8 X 12 - ELECTRONICS
POWER: 40 WATTS (10-VIDICON, 15-GIMBAL, 15-ELECTRONICS)
MTBF: 40,000 HRS.
REUSEABILITY: 50 MISSIONS
SERVICE LIFE: 2000 HRS.

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Figure 5-10. Landmark Tracker

Table 5-4. Performance Characteristics for Automatic Landmark Tracker

	DAY TIME TRACKING	NIGHT TIME TRACKING
• ILLUMINATION	SOLAR – CLEAR WEATHER (30° ELEVATION ANGLE)	1/4 MOON – CLEAR WEATHER (30° ELEVATION ANGLE)
• GROUND CONTRAST	5 : 1 (DIRT : WATER)	10 : 1 (SAND : WATER)
• IMC REQUIRED	NONE	1/2%
• CYCLE TIME PER FIX	2 SECONDS	3 SECONDS
• FIELD OF VIEW	14.3° (± 60,000 FT. FROM 80 N.M.)	14.3°
• ACCURACY	100 ARC SECONDS (WITH NARROW F.O.V. 7.2°)	
	50 ARC SECONDS	172 ARC SECONDS

5.3.4 Horizon Sensor

The system discussed herein is an improved Horizon Sensor developed by TRW Systems (Reference D-11) which is now being successfully used on the OGO Satellite. The system was developed by TRW for NASA/Goddard to fly aboard the OGO Satellite.

This Horizon Sensor system utilizes four infrared search-track units to track the earth's horizon in four planes separated 90° in azimuth. The tracking sensors measure angular elevation of the horizon in each plane from the nominal local vertical (Figure 5-11).

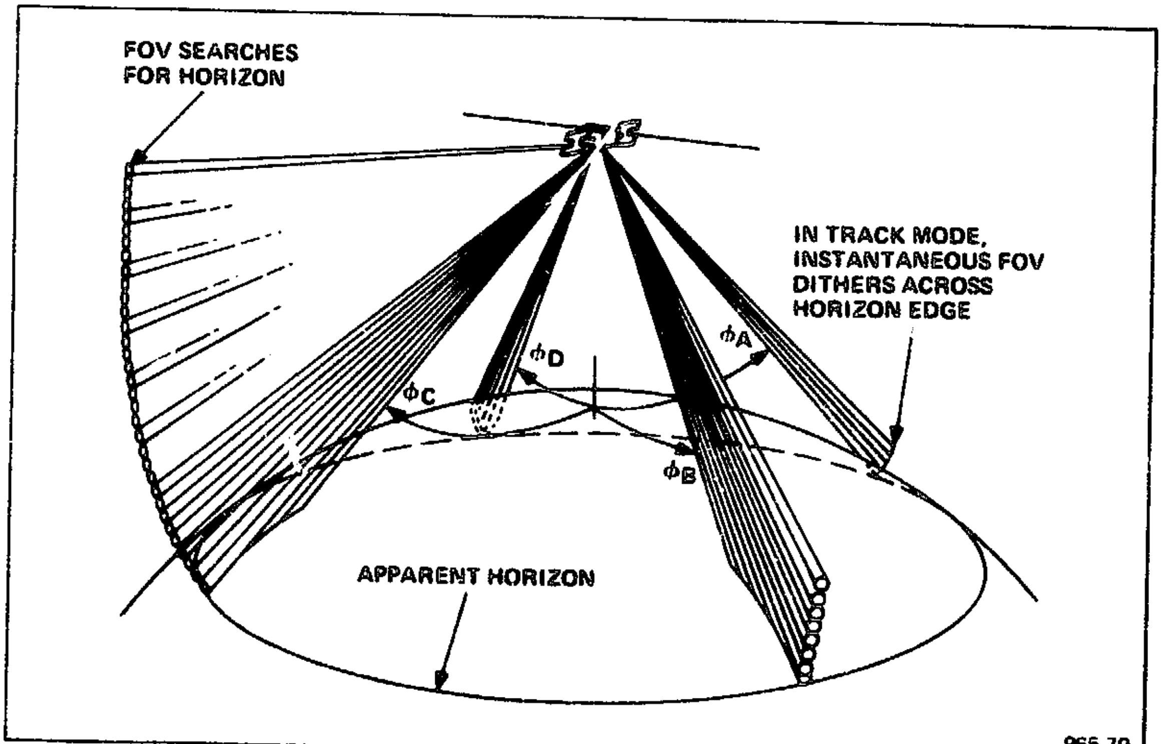


Figure 5-11. Horizon Edge Tracking Scheme

The servo circuit of each tracker includes those elements necessary to perform the search track function (telescope, thermistor, bolometer, signal preamplifier, signal amplifier, Schmitt trigger, drive amplifier, and path deflection mirror). The servo loop is intentionally made to oscillate at a constant dither frequency with a controlled amplitude. This oscillation, manifested by motion of the mirror, causes the field of view of the telescope continually to cross and recross the horizon (Figure 5-11). The output waveform of the thermistor bolometer is asymmetrical if the horizon is not in the center of the field-of-view. This results in a dc output from the Schmitt Trigger (Figure 5-12). The dc voltage drives the mirror until a null condition is reached.

The analog output of the Horizon Sensor position output is sent to a switching matrix (Figure 5-13), where logic supplied by the spacecraft computer selects the appropriate signals for local vertical computation.

The spacecraft logic bases its selection on sun presence and earth presence information from each tracker. The sun presence discrete will cause the logic to omit that tracker's signal from the matrix computation. Only three angular output signals fed via the matrix are summed by amplifiers whose output are vehicle attitude in terms of pitch and roll with respect to local vertical. The system is redundant in that the outputs of any three search-track units are sufficient to provide attitude information.

The pitch and roll attitude angles determine the local vertical of the parent planet. This local vertical orientation along with two measurements of angles to start using a star tracker, when resolved into the computational coordinates, provide the means to compute the vehicle state vector.

5.4 LANDING RADAR

A preliminary analysis to determine the requirements for a space tug landing radar has been made. The requirement for this unit and the description presented below is based primarily upon Apollo LM equipment capability. New technology promises improvements which may be ready in time for tug applications (see Section 7.5).

5.4.1 Subsystem Description

The landing radar (LR) senses velocity and altitude with respect to the lunar surface when the vehicle is moving in a near tangential approach to the lunar surface and when it rotates to a vertical attitude to complete its final descent. Velocity and altitude information is applied to the computer where it is used to check and update inertially derived data. This data is also displayed during descent from an altitude of 40,000 feet to touchdown (See Figure 5-14).

The LR is composed of an antenna assembly, electronics assembly, and a control assembly; it is functionally divided into a three-beam, continuous-wave (cw) doppler velocity sensor and a narrow-beam, linear fm/cw radar altimeter. The antenna assembly consists of a space-duplexed array of transmit and receive antennas on which the solid-state transmitters, modulator, detectors, pre-amplifiers, test modulators, and waveguides are mounted. The transmit array generates four beams, three of which are arranged in a lambda configuration and used by the doppler velocity sensor; the fourth beam is used by the radar altimeter (see Figure 5-15). The receiving antennas consist of four individual broadside

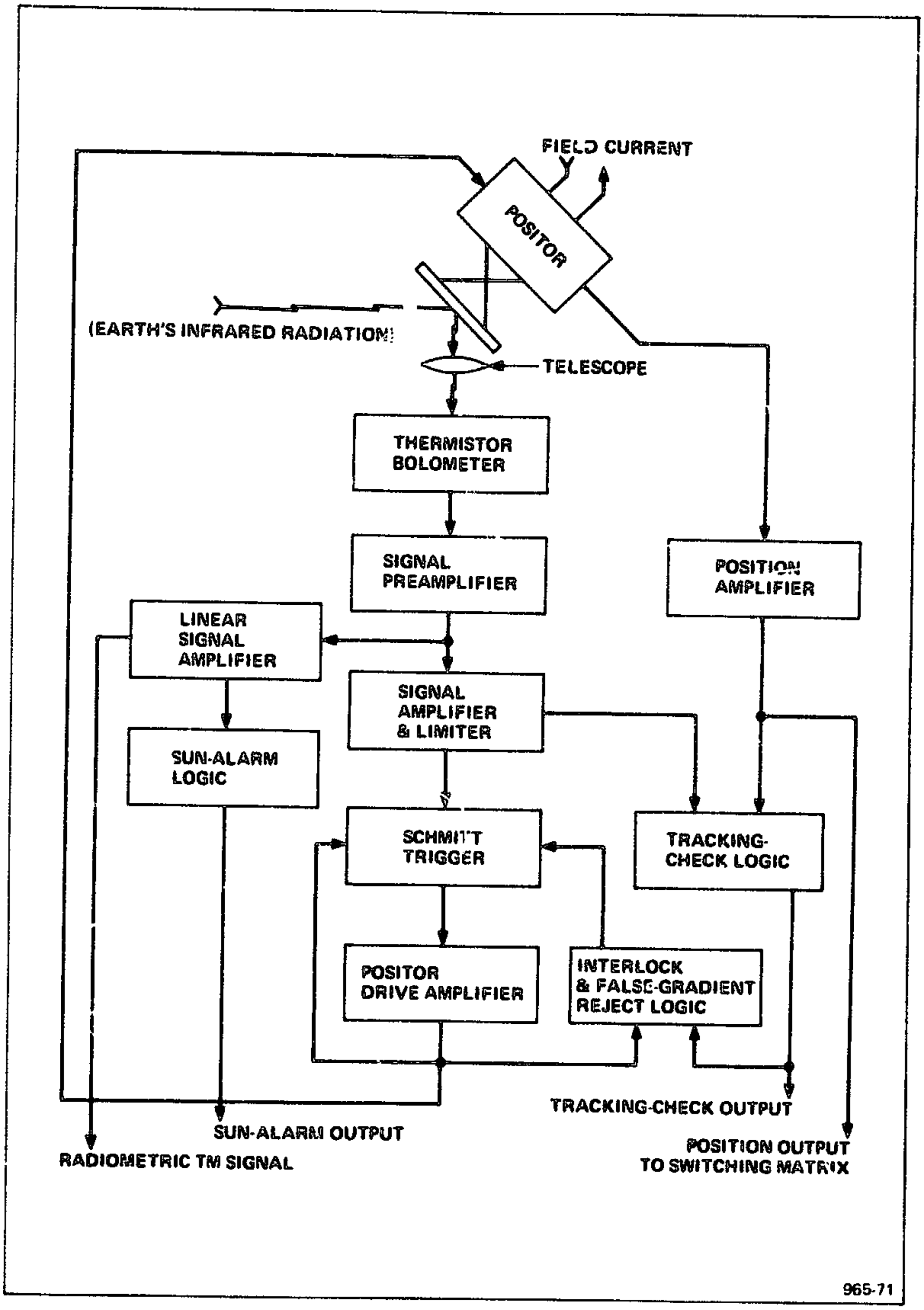


Figure 5-12. Tracker Block Diagram

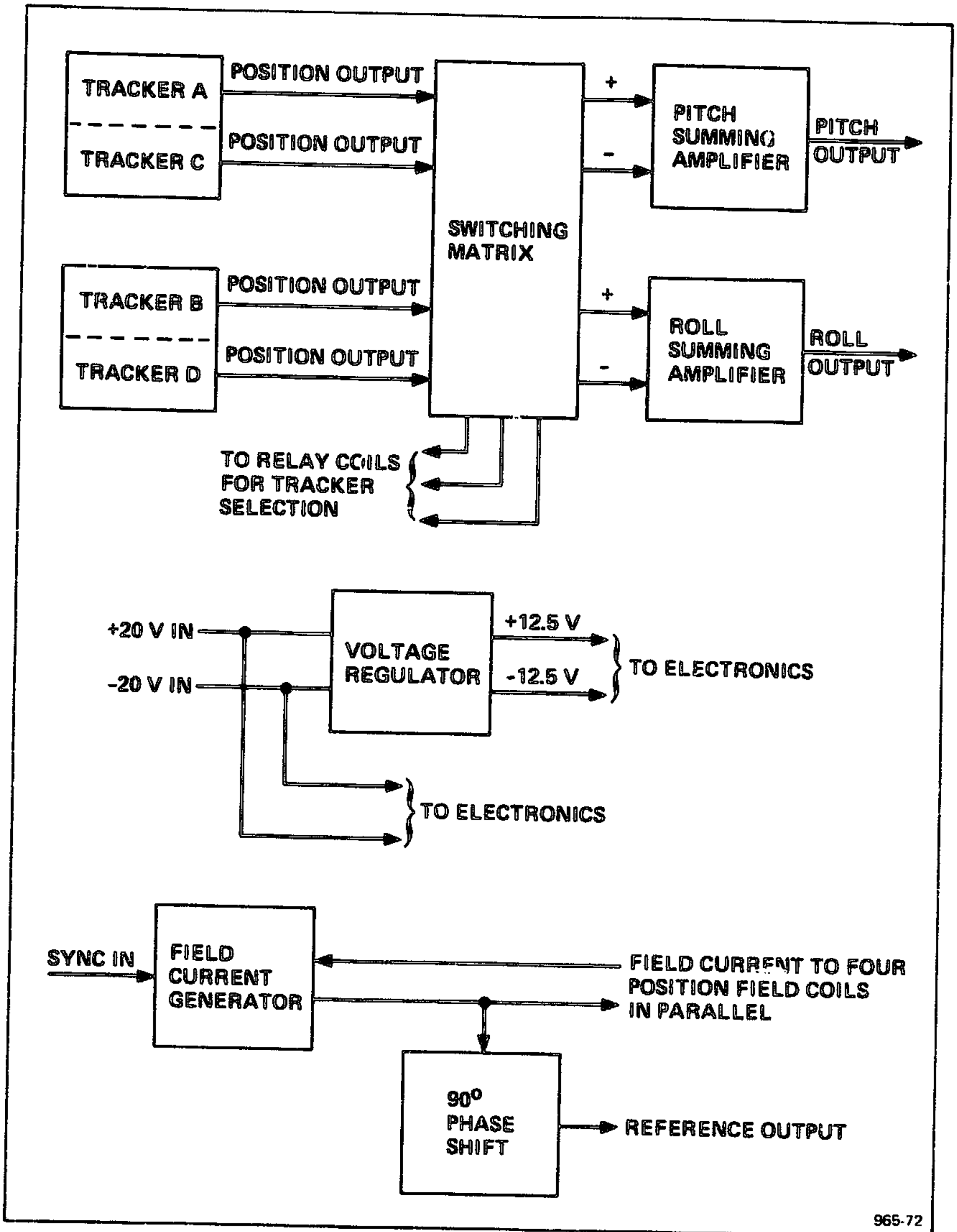


Figure 5-13. Horizon Sensor System Block Diagram

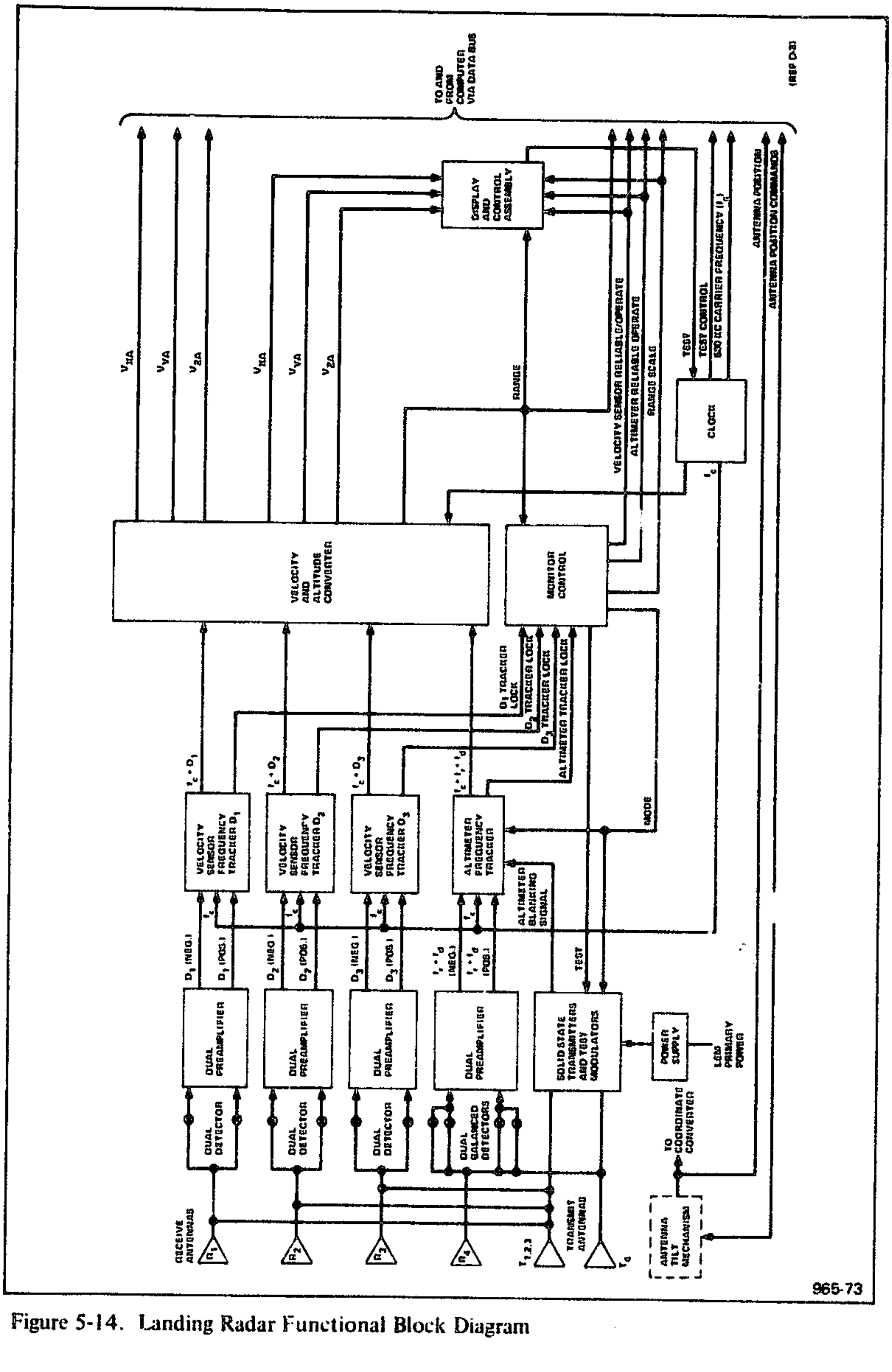
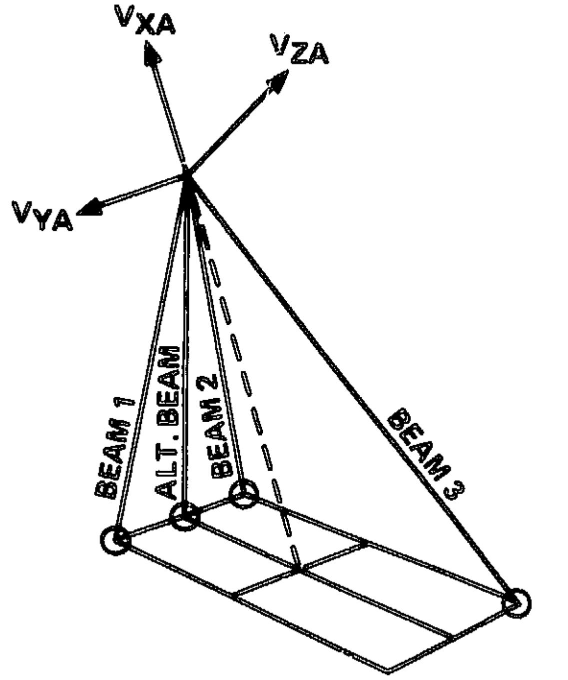
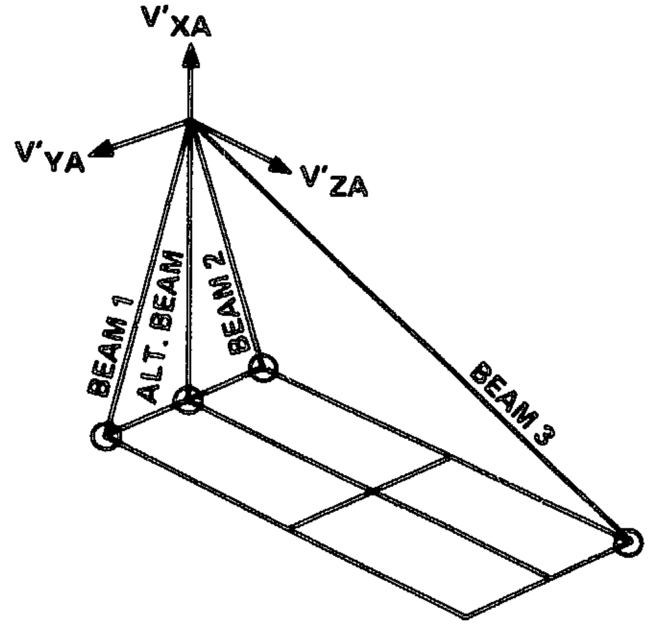


Figure 5-14. Landing Radar Functional Block Diagram



A. ORTHOGONAL SET OF VELOCITIES, V_{XA}, V_{YA}, V_{ZA} CALCULATED BY THE LANDING RADAR AND SUPPLIED TO LGC WITH V_{XA} COINCIDENT WITH CENTER LINE OF RECTANGLE.



B. ORTHOGONAL SET OF VELOCITIES V'_{XA}, V'_{YA}, V'_{ZA} SUPPLIED TO THE DISPLAYS WITH BEAM 3 V'_{XA} COINCIDENT WITH ALTIMETER BEAM.

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Figure 5-15. Landing Radar Velocity Components Relationship

arrays. The receiving array beamwidths are wider than those of the transmit array; therefore antenna boresighting is not critical. The electronic assembly contains the frequency trackers, coordinate converters, high speed counter, and the power supply. It provides binary word outputs (to the computer) that correspond to the range along the altitude beam. Pulse repetition frequency (PRF) outputs to controls and displays permit display of the velocity components (V_x, V_y, and V_z) and attitude and altitude rate. The LR supplies accurate data from 25,000 feet to touchdown without mode changes or altitude holes and has provision for hovering and negative speeds. Self-test devices within the LR enable operational checks of the entire LR without radar returns from the lunar surface; the astronauts can evaluate the operational status of the LR at any time during the mission.

5.4.1.1 Doppler Velocity Sensor

The doppler velocity sensor consists of a solid-state transmitter, frequency trackers, and beam-to-orthogonal velocity converters; it provides the desired doppler frequencies and component velocity outputs. The received energy from each beam is detected with the direct-to-audio detection technique. The received signals are detected in quadrature to retain sign sense and applied to dual pre-amplifiers. Unwanted transmitter leakage is heterodyned to zero and rejected, because the detectors are ac coupled to the pre-amplifiers. The amplified quadrature doppler signals for each beam are then applied to velocity sensor frequency trackers, which search the band of expected doppler frequencies with a narrow-band width filter. When the doppler signal appears in the tracker band, the tracker locks on and continuously tracks and filters the doppler spectrum. The outputs ($f_c + D_1$, $f_c + D_2$, $f_c + D_3$) of the velocity sensor frequency trackers are converted to V_{xa}, V_{ya}, V_{za} and applied in prf form to the high speed counter and, then, to the computer. The converter also generates output signals representing three orthogonal velocities and range rate along the altitude beam supplied in prf form to displays.

5.4.1.2 Radar Altimeter

The radar altimeter is of the narrow-beam, linear, fm/cw type; it consists of a solid-state transmitter, frequency tracker, and an altitude converter and provides outputs that represent range along the altitude beam. The received energy is detected in a manner almost identical with that of the doppler velocity sensor. The detected quadrature signals are amplified in a dual pre-amplifier and applied to the altimeter frequency tracker. The frequency along the range beam is the sum of the range frequency and the doppler frequency ($f_r + f_d$). The doppler component is removed in the altitude converter by mixing operations; the range frequency signal is applied to the altitude frequency converter, where the range signals are derived.

6.0 MOUNTING AND LOCATING CONSIDERATIONS

The following paragraphs list the more pertinent points that must be considered in mounting and locating the navigation sensors. In general, each transmitting and sensor unit must be mounted such that it has a known and fixed relationship to the primary space tug axes. In addition, each sensor must be located in a temperature controlled environment consistent with the unit operating temperature range. Typically, each sensor is designed (temperature control built in) to operate over a moderately wide ambient temperature range (0 to 50° C).

6.1 IMU

In addition to the above, the strapdown IMU should be mounted as near the center of the vehicle as practical in order to minimize the sensed centrifugal accelerations caused by vehicle rotations or on the (+) pitch axis of the vehicle if mounting in the center of the vehicle is impractical.

6.2 STAR TRACKER

The star tracker subsystems must be located to provide optical access to the stars and mounted with a known fixed orientation to the IMU. This problem was solved in Apollo by mounting both the star sensors and IMU on a rigid navigation base. This configuration permits full three-axis reference orientation measurements to a high degree of accuracy.

6.3 LANDMARK TRACKER

The sensor unit of the landmark tracker should be mounted on the rigid navigation base and located such that it has optical access to the earth or lunar surface while the star sensors are tracking stars. If this is not practical, then the sensor should be mounted on the underside of the vehicle and the sensor mount relationship to the navigation base mount should be measured.

6.4 LASER RADAR

In order to employ the laser unit for rendezvous, docking and as a landing aid, the transmitter-detector unit must be located to provide approximately 180 degree coverage. For rendezvous and docking, the sensor unit will be looking nominally along the positive longitudinal axis of the vehicle. For landing, the sensor must be capable of looking within 10 to 20 degrees of the negative longitudinal axis. The ideal location for the docking maneuvers is to have the sensor boresight axis aligned coaxially with the docking axis of the vehicle. However, if this is too great a constraint on the docking adapter design, the sensor can be located adjacent to the docking port.

The wave length of the laser is temperature sensitive. The unit as presently designed operates at 55° F $\pm 20^{\circ}$ F by using passive copper heat sinks to dissipate its own heat ($\sim$1 watt). Thus, it is necessary to regulate the temperature of the sensor such that thermal effects external to the unit not be allowed to significantly affect the sensor's internal temperature boundaries. Thermal coatings and thermos bottle insulation techniques can be used to passively maintain thermal equilibrium to within certain boundaries.

6.5 LANDING RADAR

The landing radar antenna assembly must be boresighted and located such that the transmitted signals strike the lunar surface nominally beneath the vehicle during the lunar landing phase.

6.6 HORIZON SENSOR

The Horizon sensor must have an unobstructed view of the infrared horizon. Therefore, the general location of the horizon sensor is the positive or negative yaw axis of the vehicle. Previous projects have selected the top side of the vehicle (NEG. YAW). However, the Horizon Sensor application is limited to high altitude orbit missions where the vehicle skin may obstruct the view of the earth. Therefore, it is recommended that the horizon sensor be located on the underside (POS. YAW) axis of the vehicle to permit an unobstructed view of the horizon.

7.0 FUTURE TECHNOLOGY CONSIDERATIONS

7.1 IMU

Characteristically, the strapdown and gimballed platforms differ primarily in the implementation of an inertial reference frame for accumulating specific force (gravity, thrust) measurements.

The inertial platform mechanization stabilizes the instrument package from vehicle motion through the use of gimbals. The result is a relatively passive instrument environment in which the accelerometer information can be resolved in inertial navigation data.

In contrast, the strapdown system is mounted directly on the vehicle. The sensor measurements represent specific force and angular rates in vehicle coordinates. Thus, the instruments encounter the full dynamic environment. Further, in order to maintain the inertial reference frame, an analytical computation algorithm must be implemented to resolve the instrument data into the inertial reference frame.

The strapdown IMU performance trends are better and are due principally to recent developments in permanent magnet torquers which result in more stable torquer scale factors, wheel bearing material improvement, and experiments in gas bearing gyros. Future developments in the 70's of the electrostatic and laser gyro promise to eliminate instrument moving parts with the ability to sense high vehicle rates with low drift.

Weight reduction trends in gimballed platforms are due to the use of beryllium in platforms and gimballed structures and to reduction in size of instrument sensors. Strapdown weight reduction is due principally to the elimination of gimbals and their high power requirements. In addition, the strapdown IMU lends itself to integrated packaging concepts that tend to reduce weight.

Increased gimbal platform reliability is being achieved through the use of redundant or backup systems. A disadvantage of this scheme is the large increases in power and weight.

The use of multiple gyro and accelerometer configurations in strapdown IMU's has greatly increased reliability with a modest increase in weight. The MIT hexad strapdown which contains 6 gyros and 6 accelerometers in an optimum configuration promises to provide reliability greater than a triple redundant platform configuration.

7.2 STAR TRACKERS

Star trackers are used to measure the relationship of the IMU with respect to inertial space by measuring the angular positions of the inertial components to known star positions. At present, most star trackers employ gimbals to vary their line-of-site (LOS) to known stars. Future advances will eliminate the use of gimbals and replace them with an electromechanical beam deflection scheme. This will result in significant weight reductions in the order of 10 lbs.

The recent development of completely Solid State Star Trackers promise high reliability, relative insensitivity to background light, elimination of high voltage power supplies, the use of multiple sensors, and compatibility with integrated circuit technology.

7.3 HORIZON SENSORS

There are four principal types of Horizon sensors:

1. Radiation Balance
2. Conical Scan
3. Radially oriented
4. Edge tracking

At present, only the edge tracking scheme has been widely implemented.

The horizon sensor accuracy is limited in its ability to view a definable horizon. At present that accuracy is in the order of 0.02 degrees. Future advances will not appreciably reduce this uncertainty. One horizon sensor manufactur[illegible] already uses the beam scan technique with four sensor heads. Advanced studies, pr[illegible]y underway, on fully digital horizon sensors promise future reductions in size, weight and power. In addition, the digital horizon sensor will have better maintainability, reliability and added compatibility to modularity concepts.

7.4 LANDMARK TRACKERS

The present use of the vidicon optical measurement scheme provides accurate local vertical tracking but does not provide all weather capability. In addition, the present landmark trackers use gimbals to freely rotate the sensor heads. A beam deflection mechanism will probably be implemented to replace the gimbals with a subsequent reduction in weight. Recent experiments with radiometers indicate an all weather capability but requires a 2 foot antenna operating at 15 GHZ.

7.5 LANDING RADAR

Future advances in Yagi Lasers promises to replace the present landing radars for future space missions. The Yagi Laser radar would perform the same task more accurately with less power, size, and weight. The present state of development of Yagi Lasers indicate that this equipment may be available in 1973-1978 time frame.

7.6 LASER RENDEZVOUS RADAR

The present ITT laser scanning radar is limited in power because the detector (Gallium Arsenide) is very inefficient. This power limitation limits the available tracking range to 75 nm. Experiments with other types of detectors indicate that the efficiency can be increased about 40 times. This will result in large output power thereby increasing the non-cooperative and cooperative target tracking ranges. Systems using these efficient detectors can be incorporated into the laser system by the '72 to '73 time period.

7.7 POSSIBLE COMBINATION OF NAVIGATION FUNCTIONS

It is quite likely that space communication, rendezvous and docking, and lunar landing can be accomplished using a multiple beam Yagi Laser. This sensor would have less weight, size, power, and increased reliability over the present equipment combinations.

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APPENDIX E

ANAI YSIS OF SPACE TUG GUIDANCE FUNCTION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

This appendix presents the results of the analysis of the tug guidance requirements. The approach has been to analyze a set of mission phases instead of a specific mission. This set of mission phases has been chosen such that a subset of it can be used to accomplish any of the tug missions. The mission phases addressed are tabulated below:

- Docking/Undocking
- Rendezvous (earth and lunar orbit operations and sync orbit missions)
- High Energy Maneuvers (RNS, unmanned planetary, and four stage Saturn V)
- Lunar Ascent/Descent (lunar landing mission)
- Lunar Orbit Insertion Plane Changes (four stage Saturn V or RNS)

2.0 STUDY GUIDELINES AND GROUNDRULES

2.1 DOCKING/UNDOCKING

The responsibility of guidance during the docking phase is to take the tug, at a range of approximately 1000 meters and range rate of less than 1.5 m/sec with respect to the target vehicle, and accomplish the following:

- Perform maneuvers to obtain the desired closing geometry
- Close with target vehicle such that docking is accomplished with terminal conditions within prescribed limits.

Docking can take place in either the manual or automatic mode. It is assumed that for automatic docking, the target vehicle is cooperative in that it can either be turned to a desired docking attitude or that its orientation can be determined from laser radar measurements. It is also assumed that the tug's motion is essentially coplanar with that of the target vehicle. The terminal conditions of the docking maneuver should be within the following limits (Reference E-5):

- Range rate: 1.0 ft/sec
- Lateral rate: 0.5 ft/sec
- Lateral displacement: 1.0 ft.
- Attitude error: 4.0 deg.
- Attitude rates: 0.5 deg/sec.

Radar measurements are assumed to provide range, range rate and target vehicle azimuth, elevation, and attitude.

During undocking, the guidance function is responsible for guiding the tug to a distance of 500 meters from the mating vehicle.

2.2 RENDEZVOUS

The responsibility of guidance during the rendezvous phase is to perform the targeting and issue engine start/stop and attitude commands to guide the tug from one orbit to a coplanar orbit with the target vehicle and within a range of 1000 meters with a range rate of less than 1.5 m/sec. This includes earth/lunar orbital operations and synchronous missions. It is assumed that knowledge of the target vehicle state will be provided by the navigation function. This knowledge will be in the form of a ground or space element (i.e., space station) updatable ephemeris which can be augmented by rendezvous radar.

2.3 HIGH ENERGY MANEUVERS

Included in this set of maneuvers are the injection burns for lunar and interplanetary trajectories, burns coming into and going out of lunar orbit and four stage Saturn V boost burns. The purpose of the guidance function during these burns is to place the vehicle in the desired trajectory with respect to the primary body. Targeting for lunar and interplanetary trajectories will be assumed to be similar to that used on the Apollo lunar flights. That is, the target conditions will be the elements on a conic which has been predetermined to yield the desired n-body trajectory.

2.4 LUNAR DESCENT AND ASCENT

In the lunar descent phase, the guidance scheme is responsible for issuing engine start/stop/throttle and attitude commands to guide the tug from a parking orbit to a predetermined landing sight. The guidance responsibility during the ascent phase is to issue engine commands that will take the tug from the landing sight and rendezvous with the LOSS. It is assumed that the navigation function (including landing radar) will provide position and velocity with sufficient accuracy to ensure mission success. (See Appendices B and D.) The lunar descent phase is the only tug mission phase that requires that the tug main engine be capable of throttling.

2.5 LUNAR ORBIT INSERTION PLANE CHANGES

The lunar orbit plane change analysis performed here assumes impulsive type velocity increments. That is, it is assumed that the engine burn duration is short enough to assume velocity is gained instantaneously. It is also assumed that the line of apsides of the incoming trajectory about the moon is nearly colinear with the line of nodes between the incoming trajectory and the LOSS.

3.0 SUMMARY OF RESULTS

3.1 INTRODUCTION

A summary of the guidance function requirements is tabulated in Table 3-1. As will be noted, both guidance and targeting schemes were considered. The targeting provides the guidance scheme with the end conditions it must satisfy. Targeting inputs from external sources (ground control, space station, etc.) do not effect the autonomy of the tug. In most cases targeting would be supplied before mission initiation and no other inputs would be required later, assuming on-board navigation sensors inputs were available. An exception to this is the lunar ascent where targeting information must be sent down from a space resident element.

Table 3-1. Space Tug Guidance Function Requirements

MISSION PHASE	FUNCTION	SCHEME	REQUIRED STORAGE WORDS	EQUIVALENT ADD OPERATIONS PER SEC
DOCKING/UNDOCKING	TARGETING	INPUTS FROM RENDEZVOUS RADAR	20	– –
	GUIDANCE	PROPORTIONAL $R \cdot \dot{R}$ AND $\varphi - \dot{\varphi}$	500	2100 from 1000 to 100 meters 10,500 from 100 to 0 meters
RENDEZVOUS	TARGETING	PREDETERMINED TARGET EPHEMERIS INPUTS FROM RENDEZVOUS RADAR	700	– –
	GUIDANCE	OPGUID VELOCITY DEFICIENCY IGM	3000 1000 1300	12,500 4000 2000
HIGH ENERGY MANEUVERS	TARGETING	PRECALCULATED CONIC	20	– –
	GUIDANCE	OPGUID VELOCITY DEFICIENCY IGM	3000 1000 1300	12,500 4000 2000
LUNAR DESCENT AND ASCENT	TARGETING	DESCENT - PRELOADED COORDINATES ASCENT - TARGET EPHEMERIS SUPPLIED BY LOSS	20	– –
	GUIDANCE	OPGUID APOLLO SCHEMES IGM	3000 500 1300	12,500 1200 2000

OPGUID is a guidance scheme which will handle all maneuvers the tug is required to make except docking. Because its commonality, flexibility, and ease of verification, OPGUID is the recommended guidance scheme for the space tug. (Reference Table 3-2 for a summary of tug guidance schemes.) This recommendation is compatible with the conclusion reached from studies on space shuttle guidance (Reference E-11). The main disadvantage of not using OPGUID is that many schemes rather than one must be programmed and verified to have full guidance capability.

3.2 DOCKING/UNDOCKING

Only one general guidance scheme was considered for docking. This was dictated for the most part by the assumption of full-on, full-off type RCS translational thrusters. The general characteristics of the scheme will be discussed below. It should be noted that the scheme can be used for either automatic or manual docking. The main emphasis was placed on the automatic mode since less previous work has been devoted to it. The docking maneuver will take place in the following steps:

- At a range of 1000 meters, it will be determined whether the target vehicle is active or passive.
- If the target vehicle is active, both vehicles will be aligned along the line-of-sight (LOS) between the two. If the mating vehicle is passive, its attitude will be determined and a "go-around" maneuver will be performed such that the LOS coincides with the target vehicle's axis containing the docking adaptor.
- Closing will then proceed by on-off RCS thrusting such that range rate is maintained as a function of range until docking is accomplished.

The undocking maneuver will take place in the following manner:

- The docking adaptor will release the tug.
- Using the RCS, the tug will depart from the mating vehicle along the LOS to a range of 500 meters.

3.3 RENDEZVOUS

The guidance schemes considered for the rendezvous maneuvers were a velocity-deficiency type and the fuel-optimal type (OPGUID). The choice of the scheme depends on whether the guidance requirements for a single tug mission phase or all tug mission phases are to be optimized. Table 3-1 contains a tabulation of potential schemes.

3.4 HIGH ENERGY MANEUVERS

The guidance schemes considered for the high energy maneuvers were the fuel-optimal type (OPGUID), the iterative guidance scheme (IGM) used in the Saturn V boost, and a velocity-deficiency type. Of these three guidance schemes, OPGUID appears to be the most attractive. IGM requires many presettings and much preflight analysis to insure its stability. This would be especially true for the mission where the tug astrionics guides the low thrust RNS. The velocity - deficiency scheme, though relatively simple, requires preflight analysis to insure near-optimum fuel consumption. Also, terminal trajectory errors tend to be

Table 3-2. Guidance Schemes

SCHEME	ADVANTAGES	DISADVANTAGES
OPGUID	• REAL-TIME TRUE OPTIMAL GUIDANCE • FLEXIBILITY OF USE WITH VARIOUS TUG MISSIONS • MINIMUM PRESETTINGS REQUIRED • EASE OF PROGRAM VERIFICATION • POTENTIAL FOR MAXIMUM COMMONALITY	• ADDITIONAL COMPUTER REQUIREMENTS IMPOSED • NOT USED IN CURRENT NASA SPACE PROGRAMS
VELOCITY DEFICIENCY	• DICTATES MINIMUM COMPUTER REQUIREMENTS • EASE OF PROGRAM VERIFICATION	• ADDITIONAL PREFLIGHT CALCULATIONS REQUIRED TO ENSURE NEAR OPTIMUM PERFORMANCE
IGM	• NEAR OPTIMUM PERFORMANCE	• PROGRAM VERIFICATION MORE EXTENSIVE • EXTENSIVE PRESETTINGS REQUIRED • OPTIMALITY DEGRADES OVER LONG BURN ARCS

somewhat larger with this scheme than the other two. On the other hand, OPGUID requires a minimum number of presettings and achieves the greatest accuracy but requires more storage and computer speed. A summary of the requirements for the candidate guidance schemes is presented in Table 3-1.

3.5 LUNAR ASCENT AND DESCENT

For the lunar ascent and descent phase, the guidance schemes considered were the present Apollo schemes, IGM, and OPGUID. The final choice was between the Apollo schemes and OPGUID, with the Apollo schemes being simpler and OPGUID offering more flexibility. See Table 3-1 for recommendations.

3.6 LUNAR ORBIT INSERTION PLANE CHANGES

The fuel penalties for the plane changes required for insertion of the tug into a lunar orbit coplanar with the LOSS are very severe when insertion and plane change are made simultaneously. It has been found that significant fuel savings can be realized, if relatively large plane changes are required, by inserting into a high apolune ellipse, performing the plane change near apolune and circularizing with the LOSS at perilune.

4.0 DETAILED ANALYSIS

4.1 GUIDANCE SCHEME DESCRIPTIONS

Since several of the guidance schemes to be considered can be used in more than one of the guidance functions, their descriptions will be presented here for later reference. Reference Table 3-2 for a summary of guidance scheme advantages/disadvantages.

4.1.1 OPGUID

OPGUID is a guidance scheme based on the Pontryagin maximum principle which yields fuel optimal trajectories satisfying up to six end condition constraints. The derivation of the equations in the scheme is presented in Reference E-3 and a program employing them is documented in Reference E-2. A brief description of the equations to be solved is presented below.

The governing differential equations for an optimum powered trajectory are as follows:

$$\ddot{\vec{r}} = \frac{\mu \vec{r}}{3} + a(t)\frac{\vec{U}}{U} \qquad (4\text{-}1)$$

$$\ddot{\vec{U}} = \vec{r}\,\frac{(3\,\mu \vec{r}^{\,T}\,\vec{U})}{r^5} - \mu\frac{\vec{U}}{r^3} \qquad (4\text{-}2)$$

where r = radius vector
U = vector in desired thrust direction
μ = gravitational constant
a(t) = thrust magnitude

It is desired to determine the initial conditions on these equations such that constraint and transversality equations are satisfied.

To generate the trajectory, initial guesses are made, with respect to some inertial system, for the two angles which locate $\vec{U}$ and the components of the vector $\vec{U}$. The initial magnitude of $\vec{U}$ is arbitrary. The total burn time, t_f, is also estimated and then equations 4-1 and 4-2 are numerically integrated from the initial time to t_f. Errors in the constraint and transversality equations are used to modify estimates of the initial conditions and the process is repeated until the errors are within predefined limits. The initial solution should require no more than 5 iterations and subsequent solutions are to two iterations.

The advantages of this scheme can be summarized as follows:

- Real-time true optimal guidance
- Mission flexibility
- Minimal presetting required

A comparative disadvantage of OPGUID is the large number of computer operations required in the numerical integration and iterative solution of equations 4-1 and 4-2. However, when compared to the overall capability of the tug computer dictated by other functional requirements, the addition required by OPGUID is not expected to be a deciding factor in sizing the tug computer.

4.1.2 IGM

The iterative guidance mode, IGM, is a closed loop, near-optimum guidance scheme based on the calculus of variations. That is, the iterative scheme computes steering commands as a function of the state of the vehicle-velocity, position, longitudinal acceleration, and gravitational acceleration – and the desired cutoff conditions. The guidance commands are updated each guidance cycle, using the updated state of the vehicle. The iterative guidance scheme is a path adaptive guidance scheme in that it will retain its near-optimum properties under all expected types and magnitudes of vehicle perturbations without any loss in accuracy at cutoff.

From Reference E-1 the general form of the commanded thrust direction is shown in Equation 4-3.

$$X = a + bt \tag{4-3}$$

where X = angle defining thrust direction
a,b = trajectory dependent constants
t = time

The constants "a" and "b" are determined by the position and velocity-to-be-gained to satisfy terminal trajectory conditions.

The terminal conditions for the iterative scheme are defined by five quantities: radial distance from the center of the earth, velocity magnitude, path angle against the local horizontal, inclination of the orbit plane to the equator, and descending node of the orbit plane relative to the launch meridian. The velocity will be forced to line in a plane defined by the inclination and descending node. The radius and path angle will also lie in the same plane.

4.1.3 Velocity Deficiency

In the velocity deficiency (Reference E-9) guidance scheme, steering commands are generated by directing the thrust vector along a velocity-to-be-gained vector. The desired present velocity is usually calculated by determining the conic velocity at the present radius such that the trajectory satisfies certain constraints. The velocity-to-be-gained is the difference between the desired and actual velocity vector. In most cases, the desired present velocity is obtained by the solution to Lambert's problem. This requires the vehicle to pass through a given aim point at a given time.

The velocity deficiency is the simplest of the guidance schemes considered. Its major shortcoming is that special care must be taken to insure that the performance of the scheme is near-optimum.

4.2 DOCKING/UNDOCKING

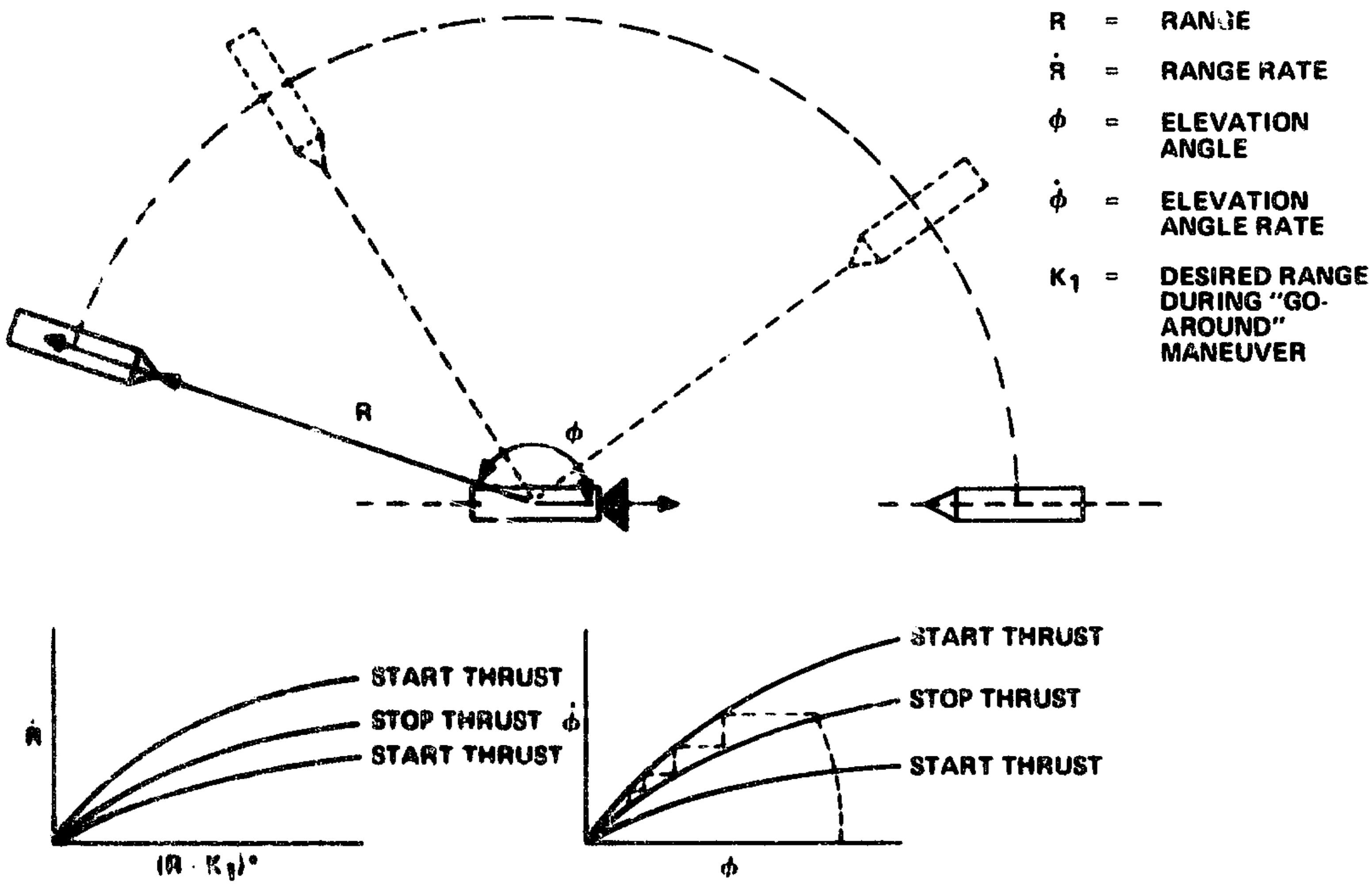
4.2.1 "Go-Around" Maneuver

Docking the tug with a target vehicle whose attitude cannot be remotely controlled presents a difficult problem. It arises because the tug must approach the target vehicle along a line through the axis of the target vehicle containing the docking adaptor. This means that the tug will, in the worst case, have to negotiate a 180 degree maneuver about the target vehicle, at a safe distance, to obtain the proper geometry for the docking to proceed. This is illustrated in Figure 4-1. Also shown graphically are the guidance laws which will control the go-around maneuver.

The range-range rate law maintains the range at a constant value, k_1, while the elevation angle is reduced to zero. The dotted line represents the trajectory of elevation angle and elevation angle rate during the maneuver. As is indicated, the tug's attitude will be maintained along the LOS during the "go-around" procedure. This phase of the docking will terminate when the elevation angle is less than 3°. The maneuver will not be performed if the elevation angle is less than 5°. During the maneuver, the azimuth or out-of-plane angle will be controlled in the same manner as the elevation angle.

4.2.2 Closing Maneuver

In the case of a passive target vehicle, the closing maneuver will be initiated by aligning the tug to the same attitude as the target vehicle. The geometry and, again, a graphical representation of the guidance laws are shown in Figure 4-2. As can be seen, the guidance laws are of the same type as for the "go-around" maneuver except that range rate is a function of unfiltered range.



* INITIAL RANGE IS 1000 METERS

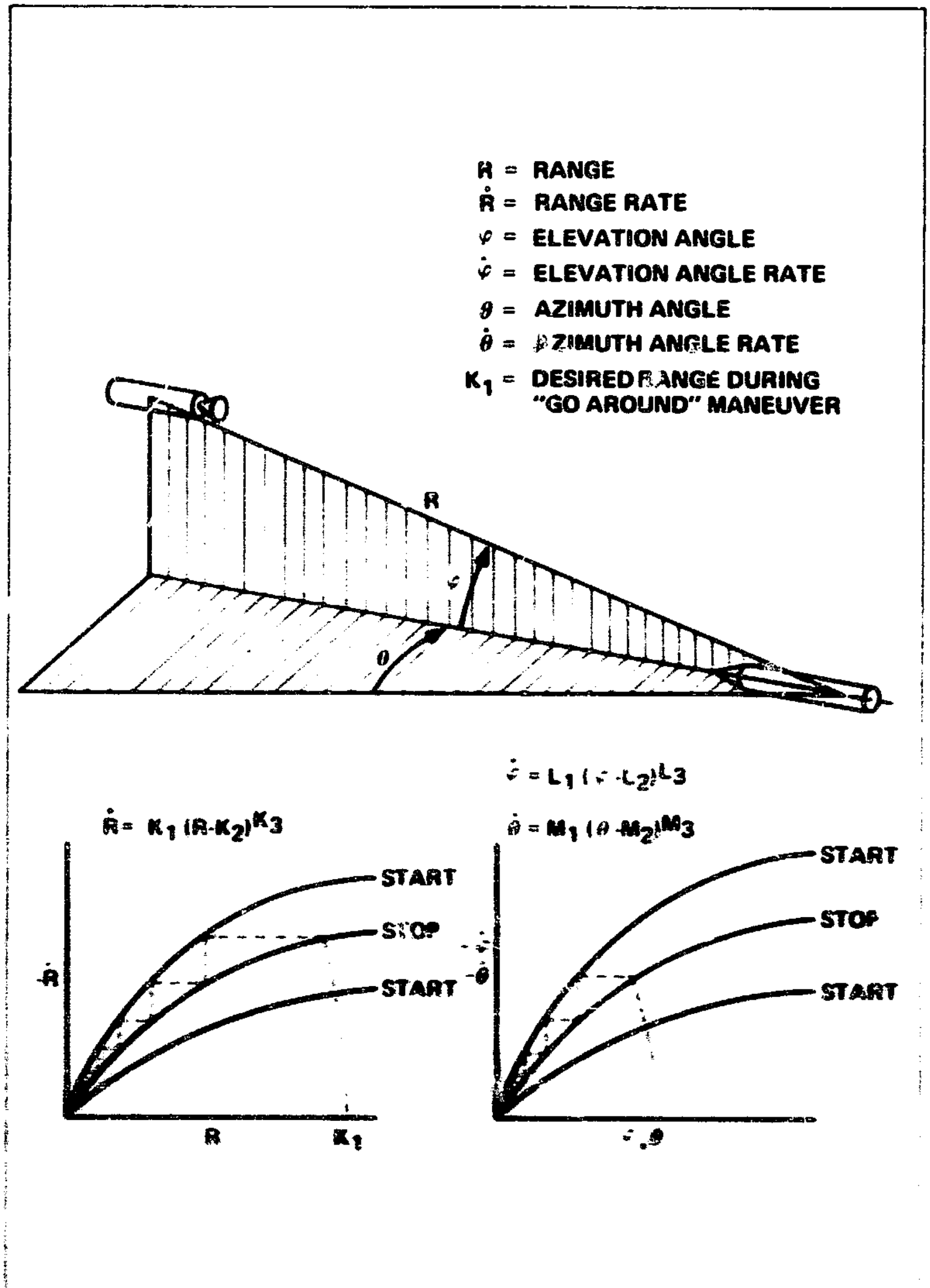


Figure 4-1 Closing Maneuver

In the case of a target vehicle whose attitude can be controlled, the "go around" maneuver and azimuth and elevation angle control are not necessary. To accomplish the docking, the vehicles would align themselves along the LOS and close with range rate as a function of range as shown previously.

4.2.3 Combination Go-Around and Closing

A procedure for reducing both the range and elevation angle simultaneously was also considered. Starting from 1000 meters, this would involve maneuvering to some intermediate range while reducing elevation. Should the elevation not be within limits when the intermediate range is reached, range will be held constant until elevation is reduced within limits and then closing will continue as described in 4.2.2.

4.2.4 Simulation Results

Simulations of the docking procedure were performed using the techniques discussed in this section. The purposes of these simulations were to:

- Confirm that the proposed docking techniques worked
- Determine the RCS impulse requirements for docking

The assumptions made in the simulation were as follows:

- Ideal attitude control
- RCS thrusters capable of a translational acceleration of 0.5 m/sec^{-2}
- Perfect range, range-rate, angle, and angle-rate information from navigation.

The guidance equations utilized were of the form shown in Figures 4-1 and 4-2. For range control, the switching curves were as follows

$$-\dot{R} = \begin{cases} 0.75\sqrt{R-K_1} \text{ (Start)} \\ 0.50\sqrt{R-K_1} \text{ (Stop)} \\ 0.25\sqrt{R-K_1} \text{ (Start)} \end{cases}$$

The elevation and azimuth switching curves were of the following form

$$-\dot{\phi} = \begin{cases} 0.0100 \sqrt{\phi} \text{ (Start)} \\ 0.0075 \sqrt{\phi} \text{ (Stop)} \\ 0.0050 \sqrt{\phi} \text{ (Start)} \end{cases}$$

$$-\dot{\theta} = \begin{cases} 0.0100 \sqrt{\theta} \text{ (Start)} \\ 0.0075 \sqrt{\theta} \text{ (Stop)} \\ 0.0050 \sqrt{\theta} \text{ (Start)} \end{cases}$$

It should be noted that deadbands about the origins of the phase plane guidance curves, equal to one-half of the allowable docking tolerances, were also simulated.

Initial conditions for each of the simulations were:

Range = 1000 m
Range rate $\simeq$ 0 m/sec
Elevation = 0 to 180°
Elevation rate $\simeq$ 0°/sec
Azimuth $\simeq$ 0°
Azimuth rate $\simeq$ 0°/sec

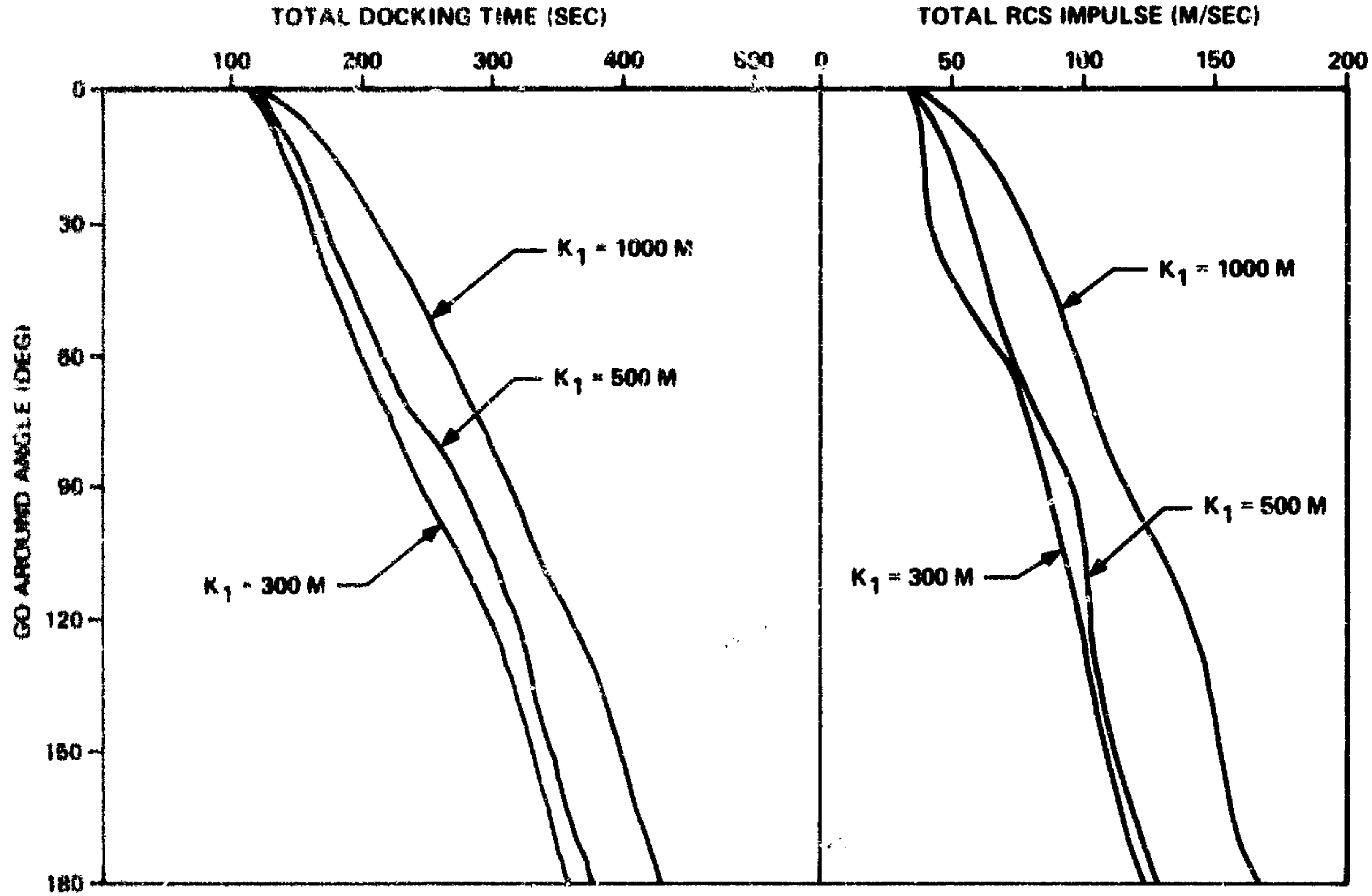
The two quantities that were varied were the initial elevation angle (ϕ) and the intermediate range the tug must maintain until the desired elevation (within limits) is achieved. A summary of the total docking time and the RCS fuel requirements is plotted in Figure 4-3. As was expected, the time and RCS fuel requirements increased as the go-around angle increased. It is also indicated that these quantities increased as the intermediate range was increased.

Figures 4-7 through 4-15 present plots of how the various quantities involved reacted during a docking simulation with a go-around angle of 180° and an intermediate range of 1000 meters. As can be seen, all quantities performed in accordance with the suggested guidance laws. Similar plots for an intermediate range of 500 meters are presented in Figures 4-16 to 4-24. The seemingly erratic behavior near the origin of some of the phase plane plots is due to the deadbands used in the simulations.

It should be noted that constants in the guidance equations used here do not represent an optimal choice. They were chosen to demonstrate that the basic techniques for docking guidance are sound.

4.3 RENDEZVOUS

The burns which are required for the rendezvous maneuver are shown in Figure 4-4. As can be seen, from four to seven burns are necessary. The first burn establishes a transfer ellipse from the initial orbit which intersects a pseudo orbit at least once. Capability should also exist for a small midcourse correction burn before intersecting the pseudo-orbit to cancel out initial burn errors. The third burn performs a plane change to establish the tug in an orbit which is coplanar with the orbit of the target vehicle. This is followed by a burn to establish an orbit with a constant differential height (10-15 km) below the target vehicle orbit. If possible, the transfer trajectory is designed such that the third and fourth burns can be made simultaneously. The fifth burn places the tug on a transfer ellipse such that it will intercept the target vehicle after a prescribed coast period. The seventh burn will place the tug in the same orbit as the target vehicle at a range of approximately 1000 meters. Provision is also made for a small correction burn between the fifth and seventh burns, should it be necessary. The guidance schemes considered for the rendezvous burns were OPGUID (see 4.1.1) and velocity deficiency (see 4.1.3).



INITIAL RANGE – 1000 M
TRANSLATIONAL ACCELERATION – 0.5 M/SEC^2
K_1 – RANGE AT WHICH CLOSING MANEUVER BEGINS

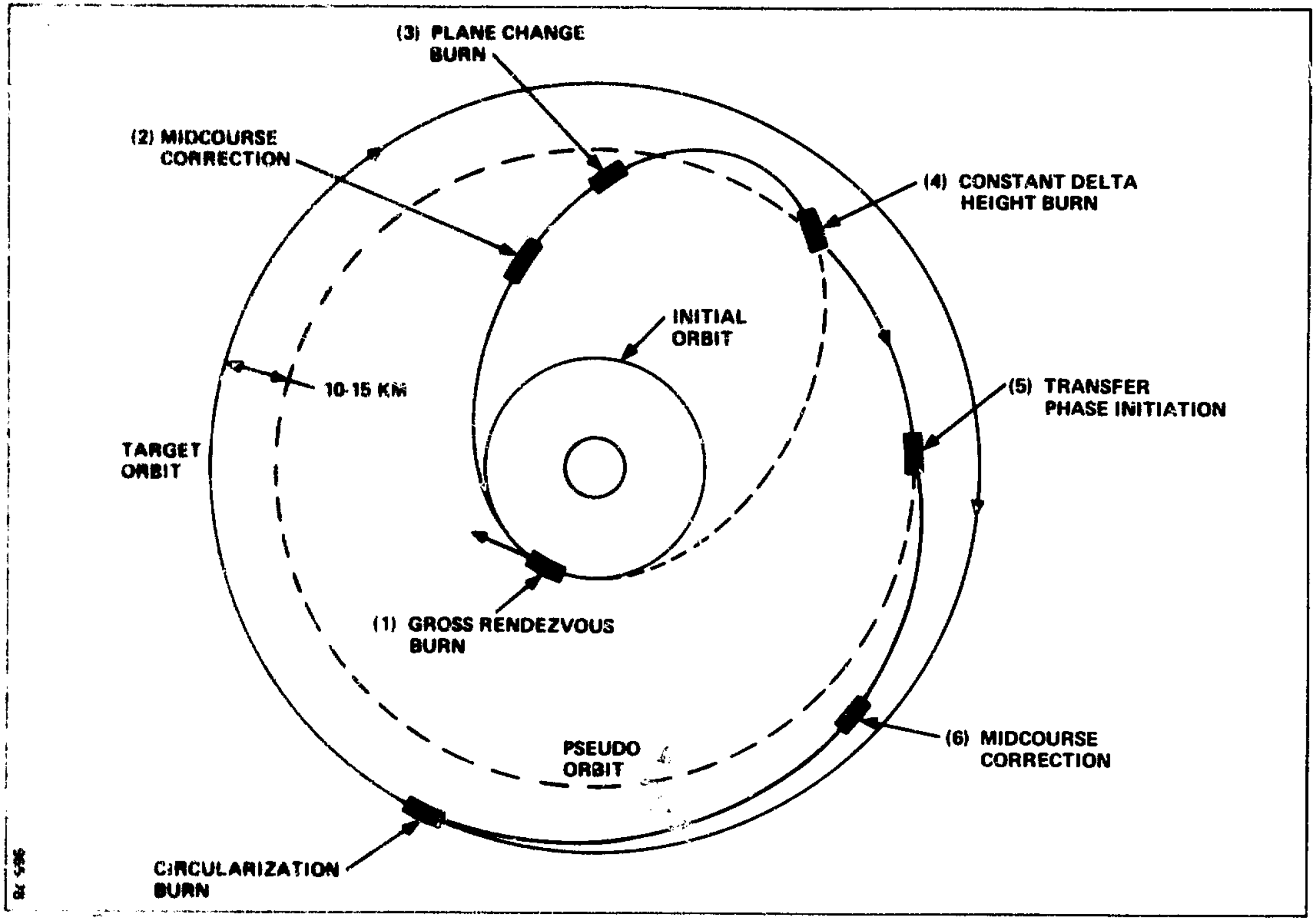


Figure 4-4 Rendezvous Geometry

The targeting for all the burns in the rendezvous procedure will be calculated using conic methods (See References E-3 and E-5). Equations for the targeting are in the process of being developed under Appendix 10 of contract NAS8-14000. It should be noted that the rendezvous techniques can be applied to any type problem from low-earth-orbit-to-synchronous-orbit type to the simple earth/lunar orbit operations type.

Once the targeting calculations have been made, guidance will be accomplished using either a velocity-deficiency or optimal guidance law. Recommendations are made in Table 3-1.

4.4 HIGH ENERGY MANEUVERS

The phases of guidance addressed here are those which are responsible for putting the vehicle on a lunar or interplanetary trajectory, for entering or leaving lunar orbit, or boosting to earth parking orbit. Since accuracy and fuel economy are particularly important, only optimum or near-optimum guidance techniques were considered. The optimization referred to is with respect to fuel in all cases.

The schemes that were considered were OPGUID, IGM, and velocity deficiency. The bulk of the detailed analysis for these maneuvers was considered with the schemes themselves. A detailed description of each scheme can be found in section 4.1.

Assuming computer requirements are not a limiting factor, OPGUID is clearly superior to the other two guidance schemes. However, if OPGUID is not used, the choice between IGM and velocity deficiency will depend on the vehicle characteristics. For short, high thrust burns where fast guidance reaction is required, IGM is superior to velocity deficiency. For low thrust, long duration burns the velocity deficiency scheme has the advantage. An example of the low thrust case would be the RNS mission (Reference E-9).

4.5 LUNAR ASCENT AND DESCENT

The following is a discussion of guidance schemes that could be employed in the ascent and descent phases of the tug lunar landing mission. The first scheme is a near-optimal solution to the problem of arriving at a given state at a specified time. The scheme can easily be applied to descent or ascent. Assuming a constant gravitational attraction, the calculus of variations solution which minimizes the integral of acceleration is (Reference E-4, page III-35)

$$\vec{a}_T = \frac{4}{t_{GO}} \left(\vec{V}_D - \vec{V} \right) + \frac{6}{t_{GO}^2} \left[\vec{r}_D - \left(\vec{r} + \vec{V}_D t_{GO} \right) \right] - \vec{g} \qquad (4-4)$$

where $\vec{a}_T$ = desired thrust vector

$\vec{V}$ = present velocity vector

$\vec{V}_D$ = desired final velocity vector

$\vec{r}$ = present radius vector

$\vec{r}_D$ = desired final radius vector

t_{GO} = time-to-go until final state

$\vec{g}$ = gravitational acceleration

It should be noted that Eq. 4-4 specifies both thrust magnitude and direction. This implies the use of a throttleable engine.

There are several weak points in this solution. First, the optimization criteria, integral of acceleration, assumes that specific impulse is constant regardless of thrust level. This is not always the case. Also, over the burn arcs that are encountered, the assumption of constant gravity is not valid. The situation can be improved by substituting an average gravity vector into Eq. 4-4. It should also be pointed out that the guidance equation tends to become unstable as "t_{GO}" approaches zero. The singularity in the equation can be avoided by holding t_{GO} to a constant value when it becomes less than a pre-assigned limit. With its weaknesses, however, Eq. 4-4 has been found to yield near-optimum solutions to a variety of problems. A slightly modified version of Eq. 4-4 is used in the Apollo lunar ascent scheme (see Reference E-4, section III).

The second candidate system is the one presently used for Apollo lunar descent. It assumes that the thrust vector can be expressed as a polynomial of the form (Reference E-8 page 2B-95).

$$\vec{a}_T = \vec{K}_1 + \vec{K}_2 t_{GO} + \vec{K}_3 t_{GO}^2 \qquad (4\text{-}5)$$

Solving the equations of motion using Eq. 4-5 yields the desired descent thrust as a function of "t_{GO}" and the desired final state (Reference E-7 page 5.3-119).

$$\vec{a}_T = \vec{a}_D - \vec{g} - \frac{6}{t_{GO}} (\vec{V}_D - \vec{V}) - \frac{12}{t_{GO}^2} (\vec{r} - \vec{r}_D) \qquad (4\text{-}6)$$

It should be noted that the above equation also implies the use of a throttleable engine.

One minor difficulty occurs using any of the suggested guidance schemes. This difficulty is that commanded attitude, at landing for descent and at liftoff for ascent, cannot be constrained. The problem is usually circumvented by dividing the powered portion of flight into two segments, a fixed attitude portion and an active guidance portion. In the ascent phase, this would mean a vertical rise for a short period and then active guidance. In the descent phase, the guidance system would place the vehicle at a given altitude over the landing sight with zero relative velocity, then a vertical descent until touchdown.

Throttling capability is a necessity for lunar landing missions. This requirement is imposed because of the following reasons:

- The vehicle must hover and make landing site adjustments.
- The final stage of landing requires a controlled rate of descent.
- Several of the candidate guidance schemes specify thrust magnitude changes.

4.6 LUNAR ORBIT INSERTION PLANE CHANGES

A tug entering lunar orbit from the earth may have to negotiate up to a 90° plane change in order to rendezvous with the lunar orbiting space station (LOSS). It is desired to perform the plane change maneuver when the vehicle velocity is as low as possible to optimize the fuel requirements. One method of accomplishing this is to make the plane change burn at the apolune of a high elliptical orbit. The geometry of the maneuver is shown in Figure 4-5. As is indicated, three burns are required. It should also be noted that this geometry assumes that the line of apsides of the incoming hyperbolic orbit coincides with the line of nodes between the incoming hyperbolic orbit and LOSS orbit.

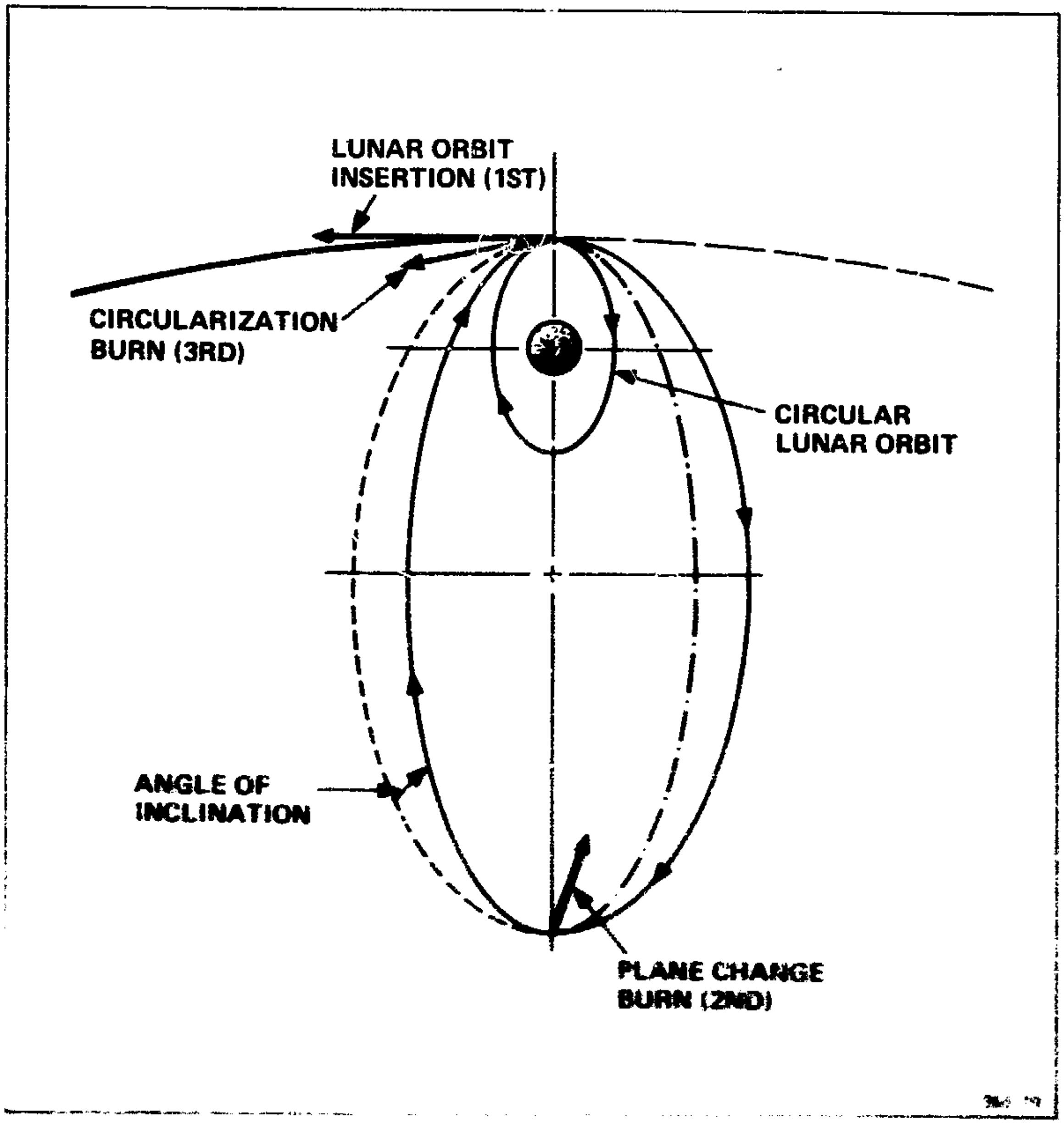


Figure 4-5. High Apolune Plane Change Maneuver

To gain some insight into how much velocity impulse the three burn maneuver can save, a standard Apollo incoming hyperbolic trajectory was chosen as a test case. The trajectory had a semi-major axis of 4222.6 kilometers and an eccentricity of 1.437. This resulted in a perilune radius (R_p) of 1845 kilometers.

A plot of velocity impulse as a function of apolune radius (R_A) and plane change was formed. The results are shown in Figure 4-6. The solid line represents the case where the plane change and circularization are accomplished in one burn. The other curves are for the three burn maneuver with apolune radii as labeled. From the plot, it is evident that the three burn mode is more efficient than a single burn when the apolune radius is greater than 8845 km and the plane change is greater than 5°.

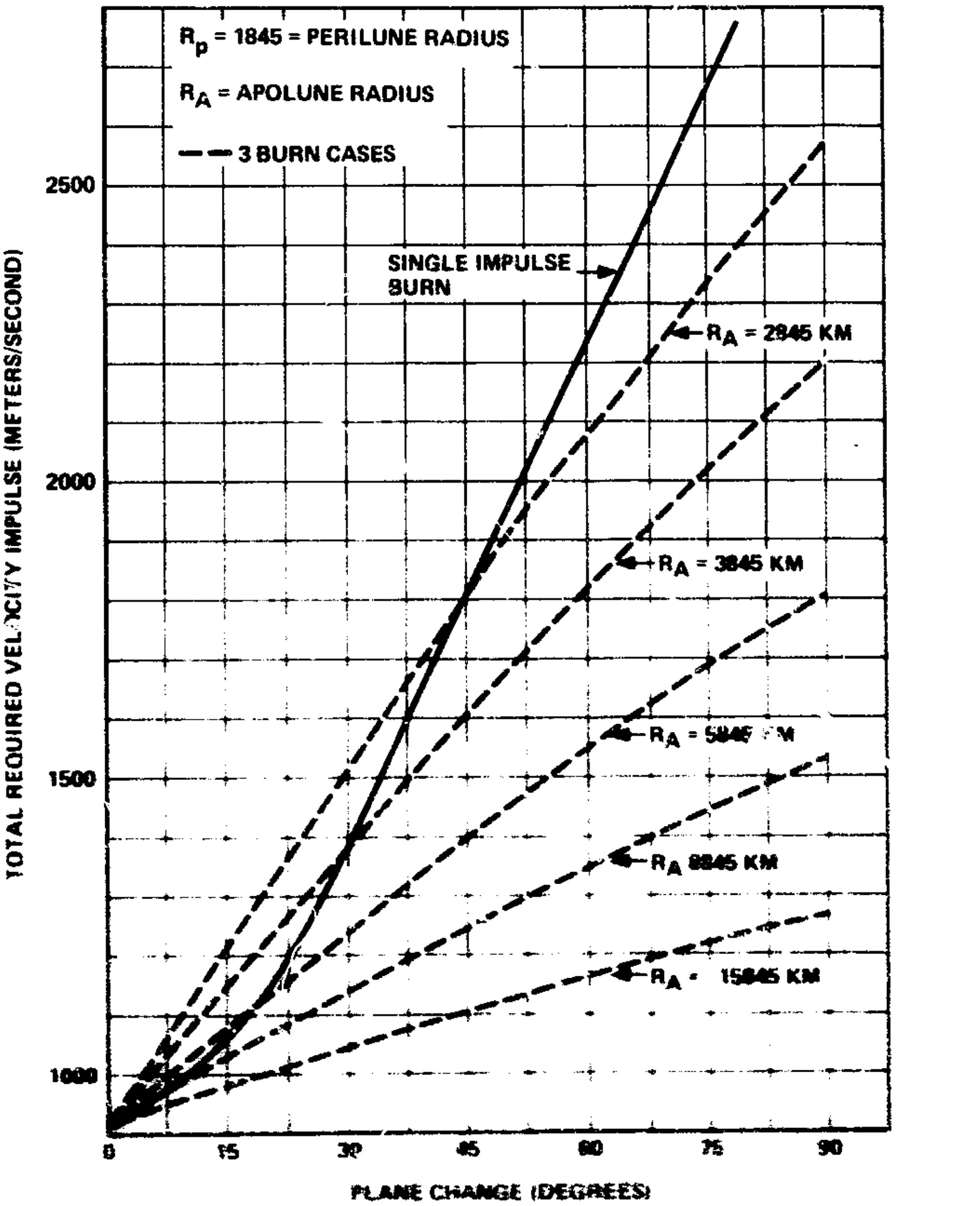


Figure 4-6 Lunar Plane Change Velocity Impulse Requirements

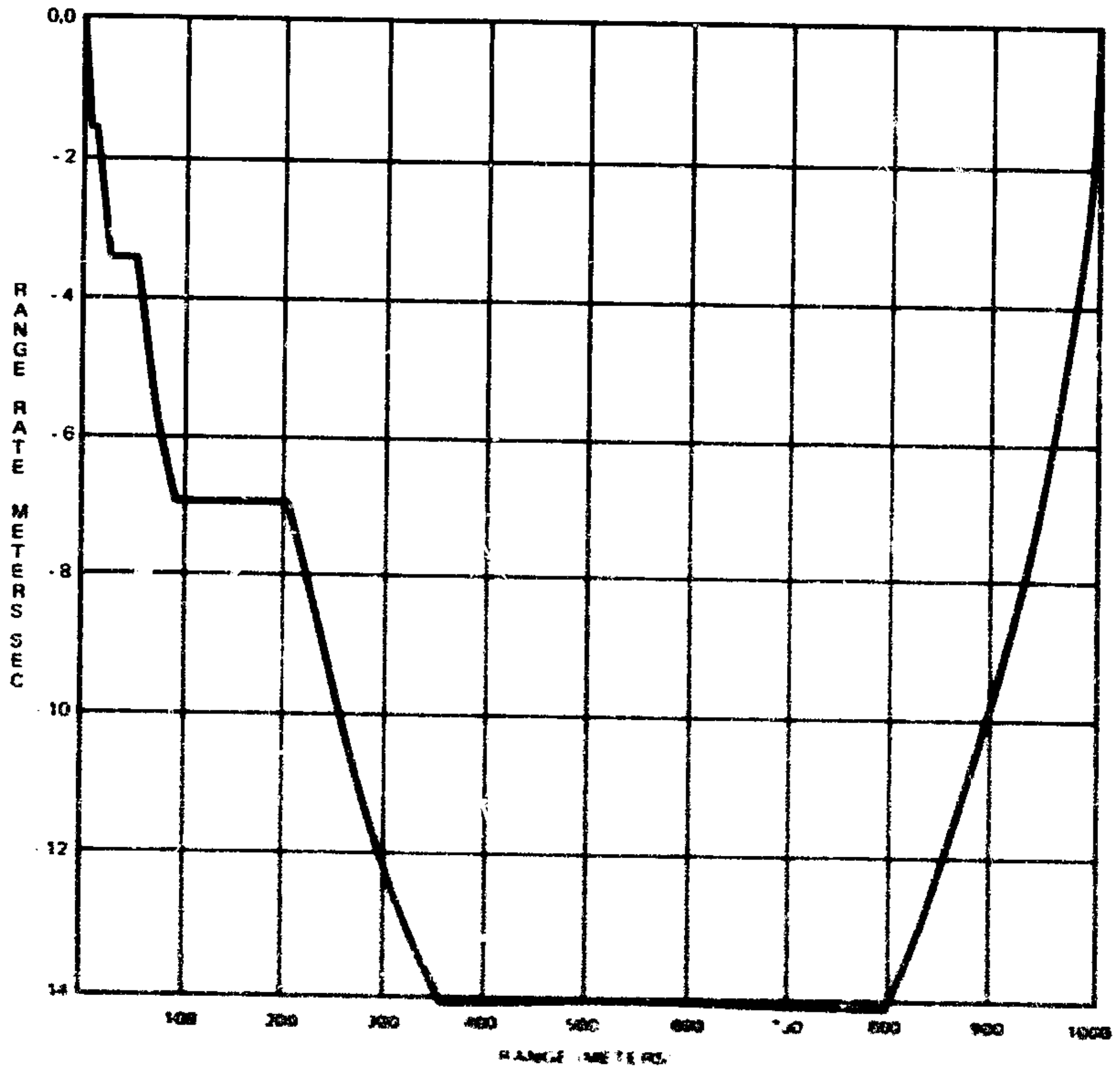


Figure 4-[illegible] Range - Range Rate Phase Plane [illegible] Range [illegible] Meters and [illegible]

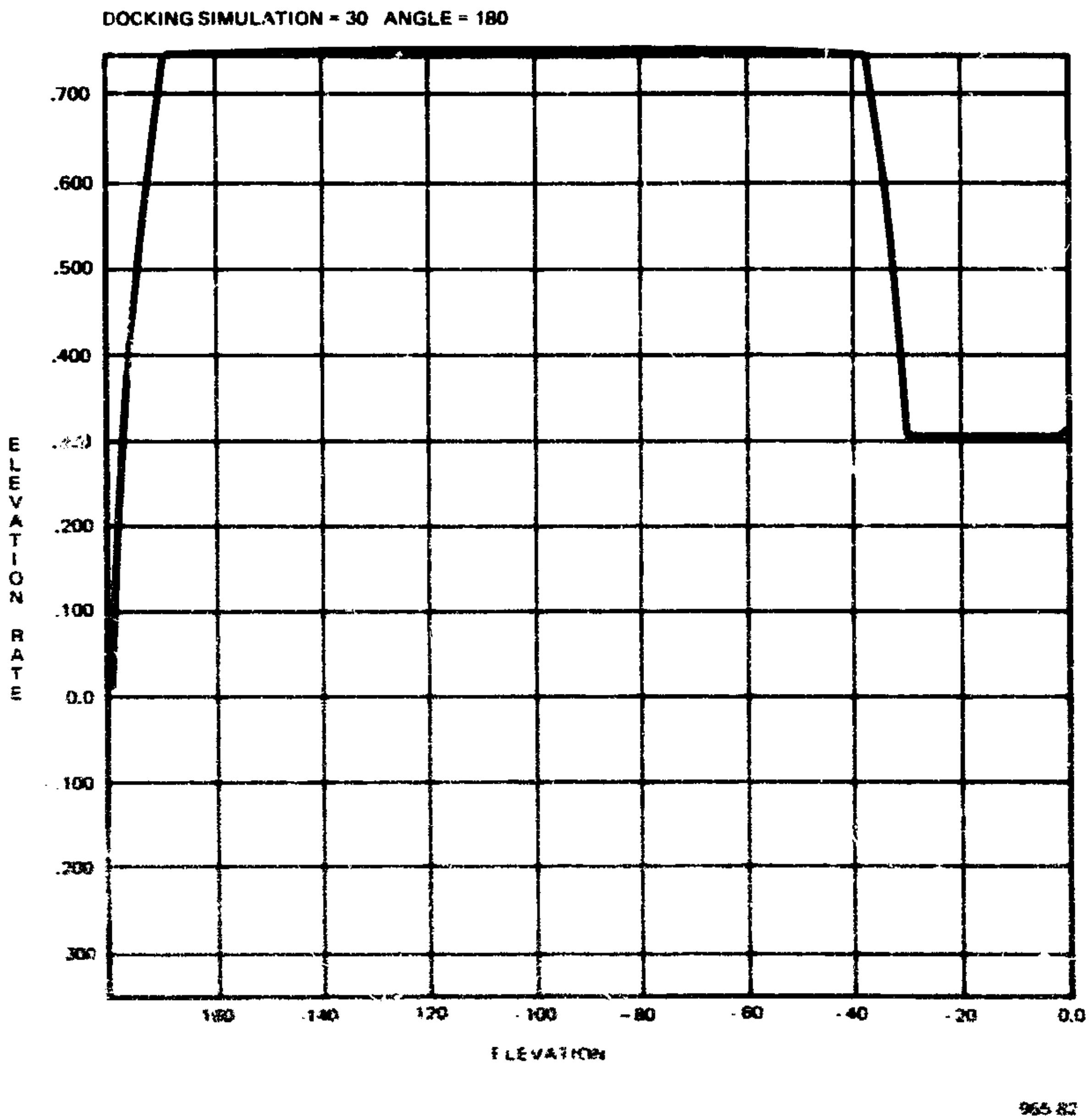


Figure 4-[illegible] Elevation–Elevation Rate Phase Plane for Intermediate Range = [illegible]000 Meters and Go-Around Angle = 180 Deg

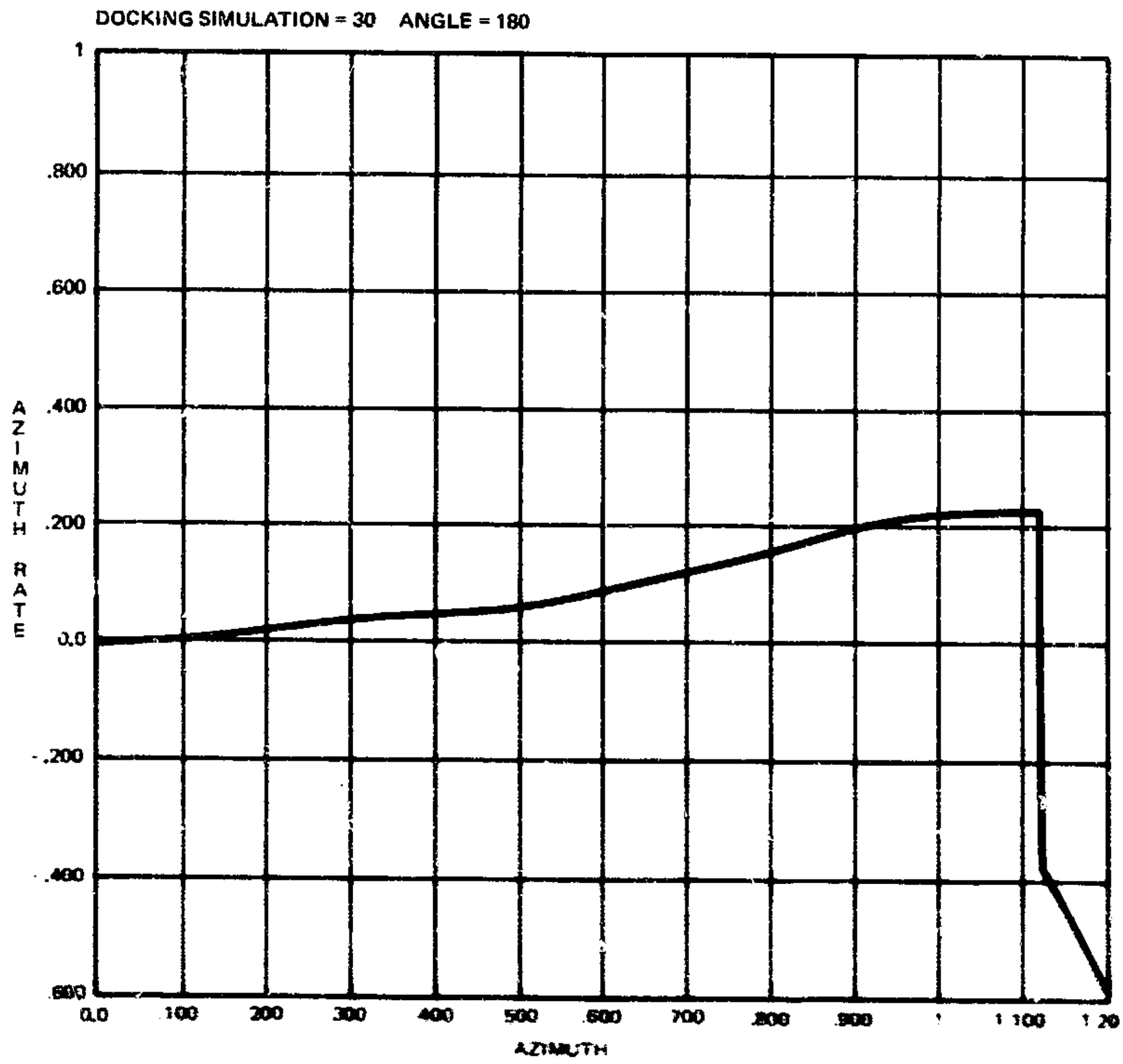


Figure 4-[illegible] Azimuth - Azimuth Rate Phase Plane for Intermediate Range [illegible] Meters and Go Around Angle [illegible]80 Deg

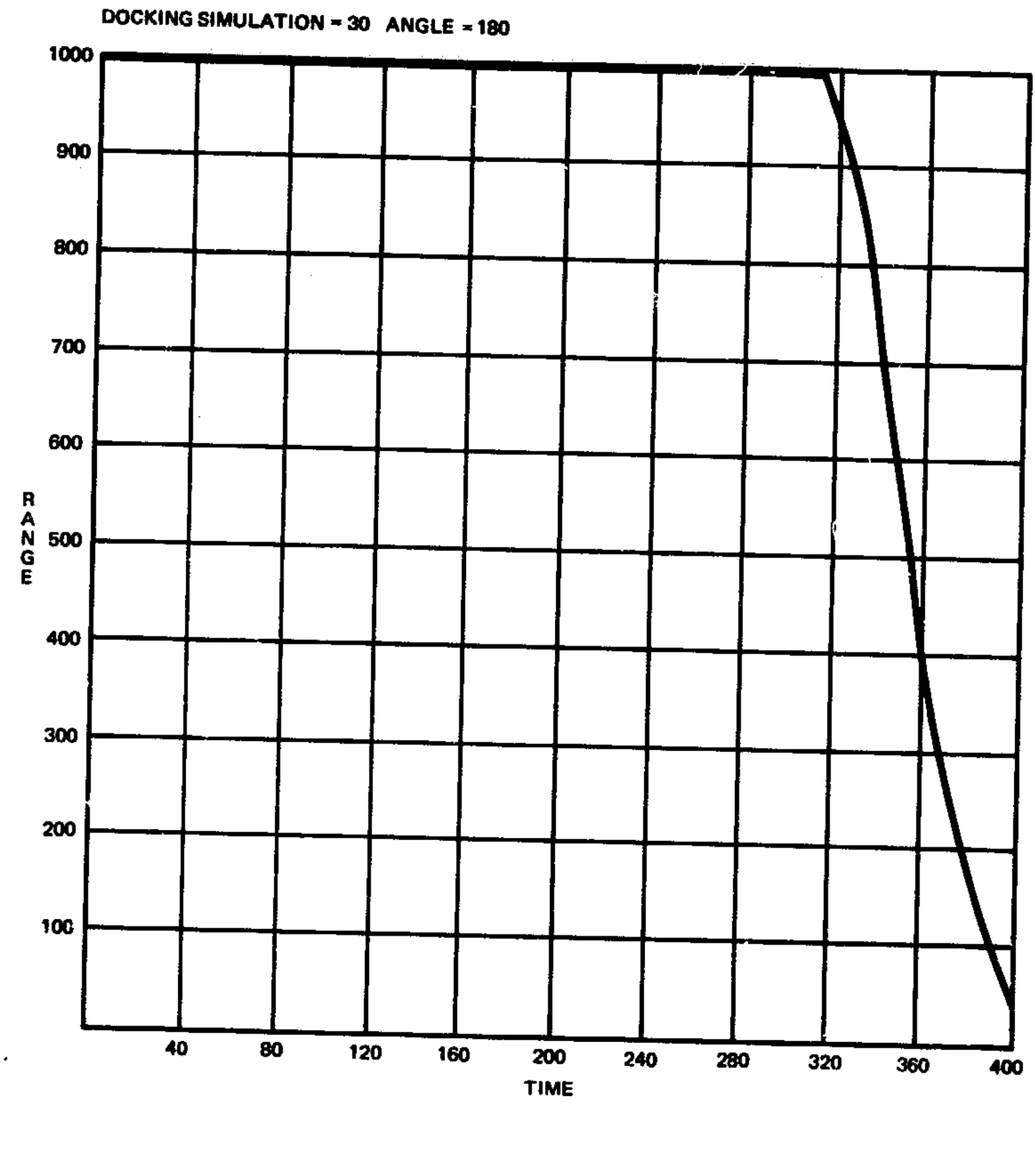


Figure 4-10. Range VS. Time for Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

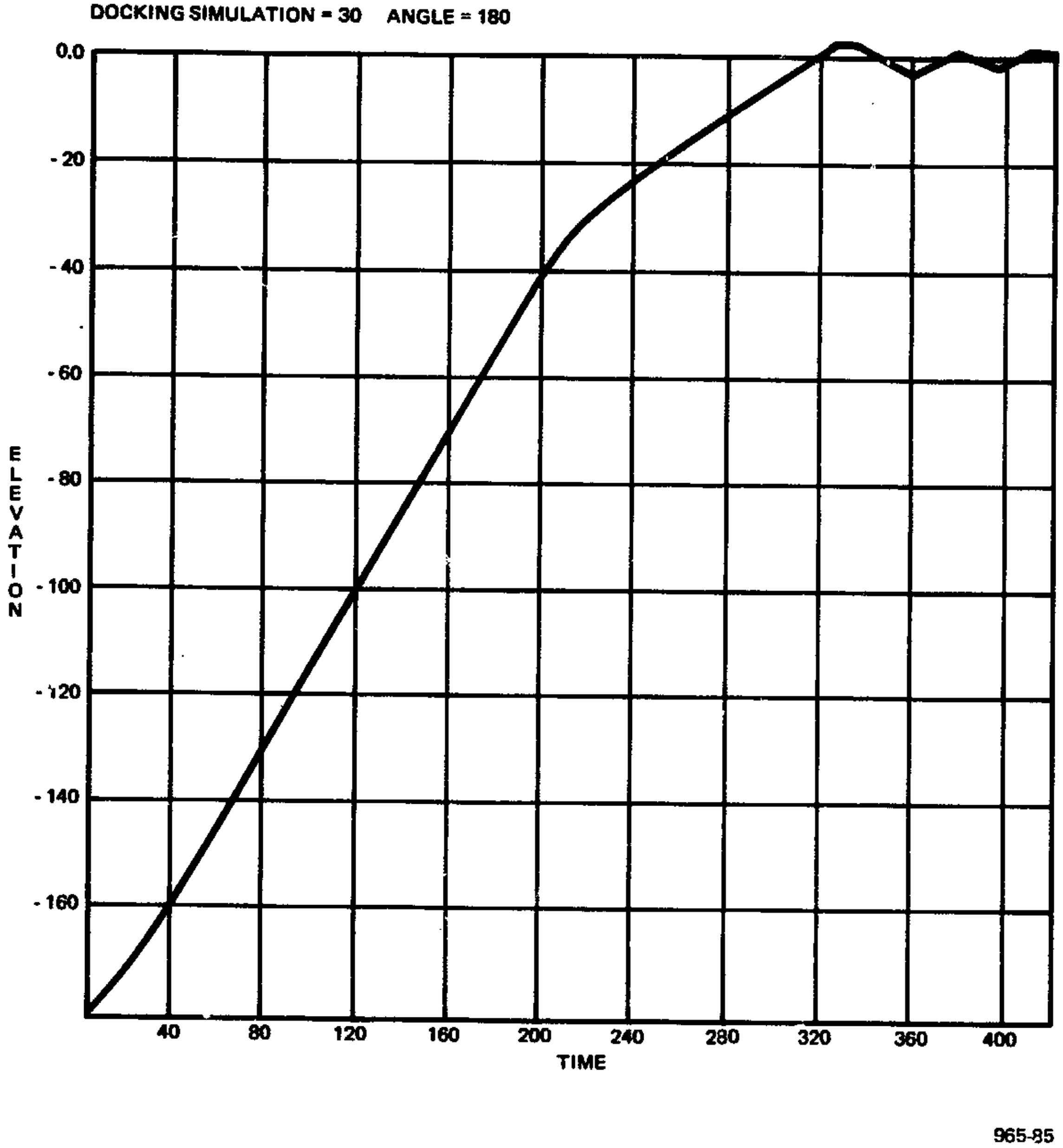


Figure 4-11. Elevation VS. Time for Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

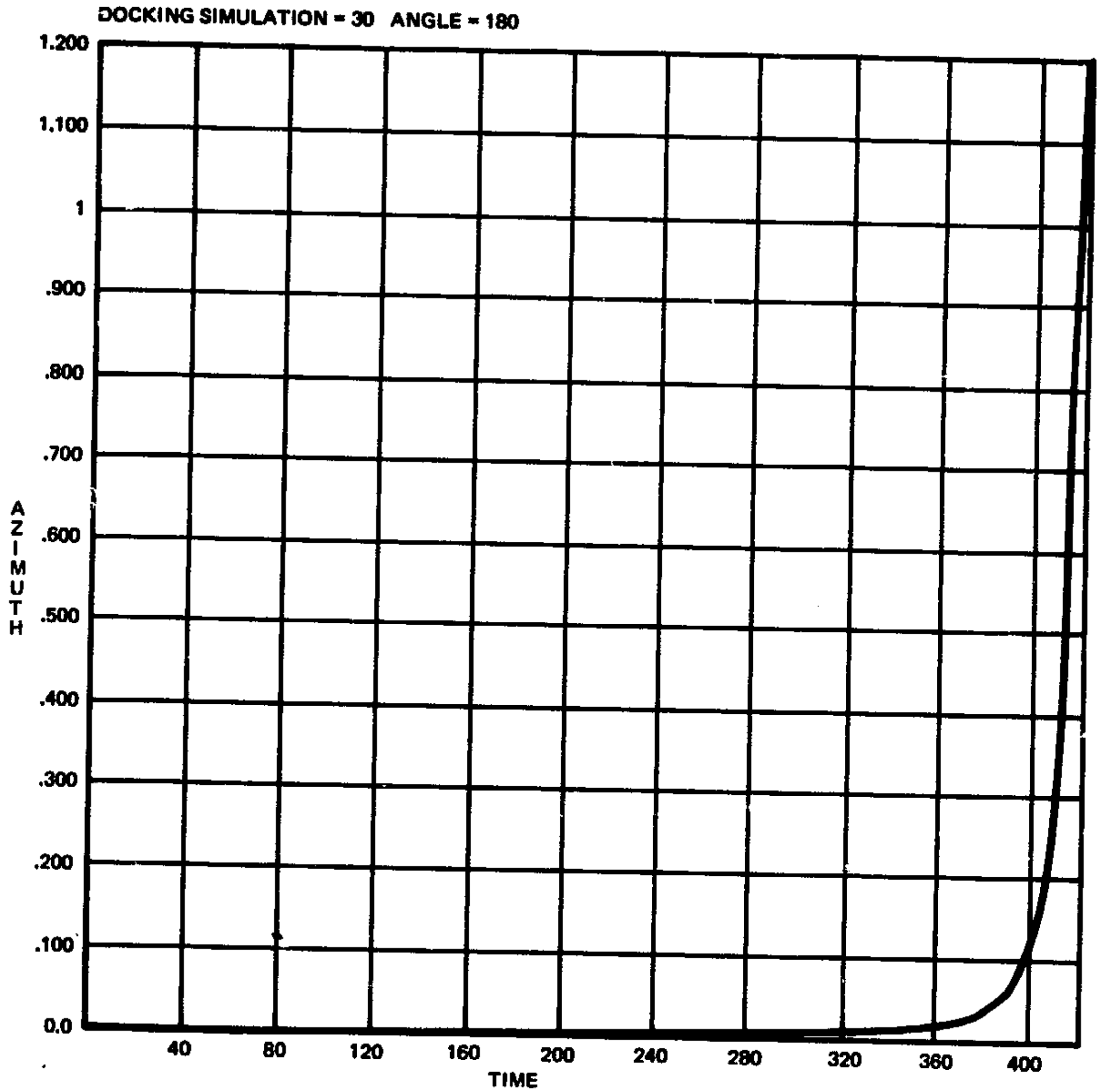


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Figure 4-12. Azimuth VS. Time For Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

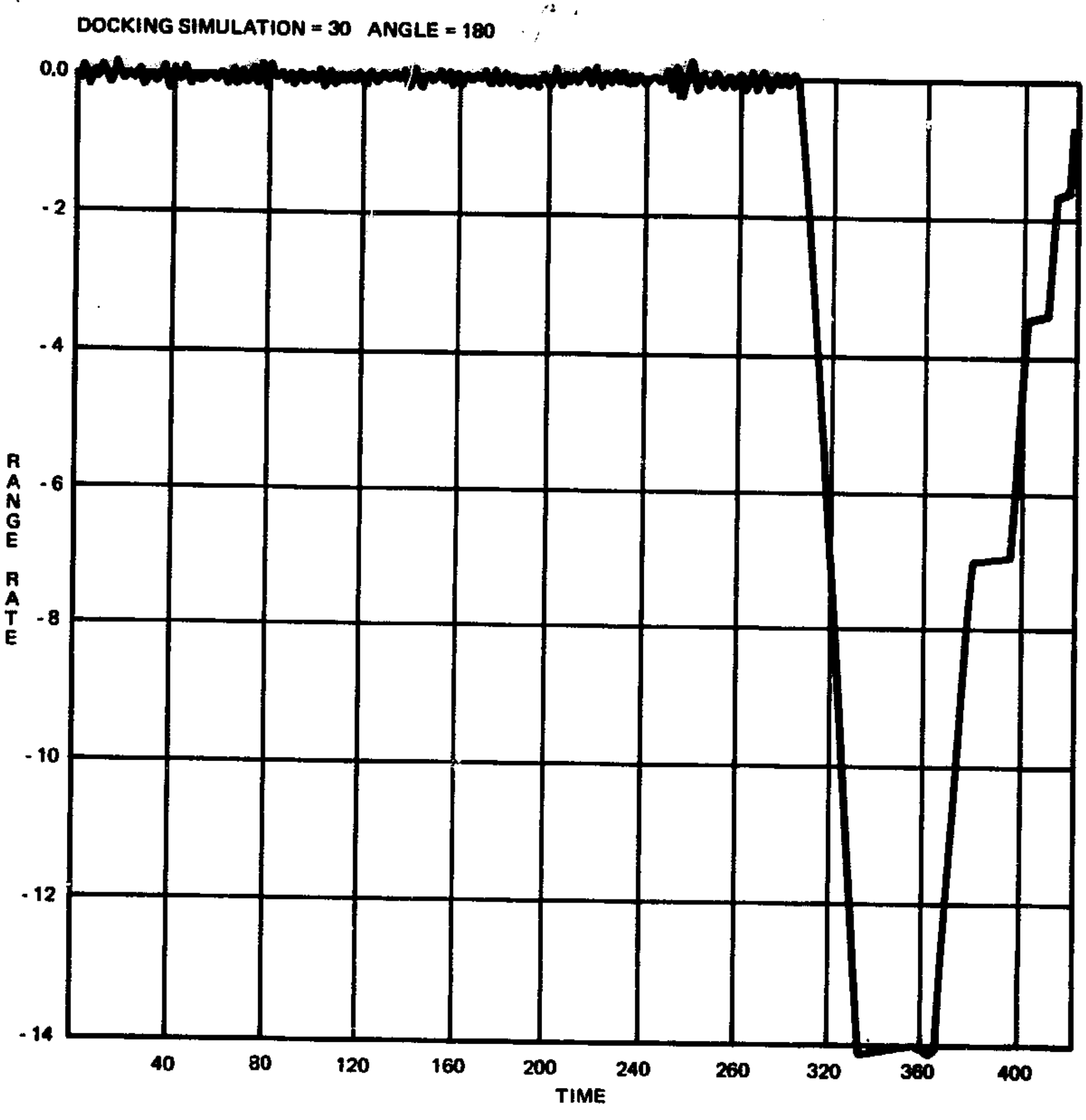

Figure 4-13. Range Rate VS. Time for Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

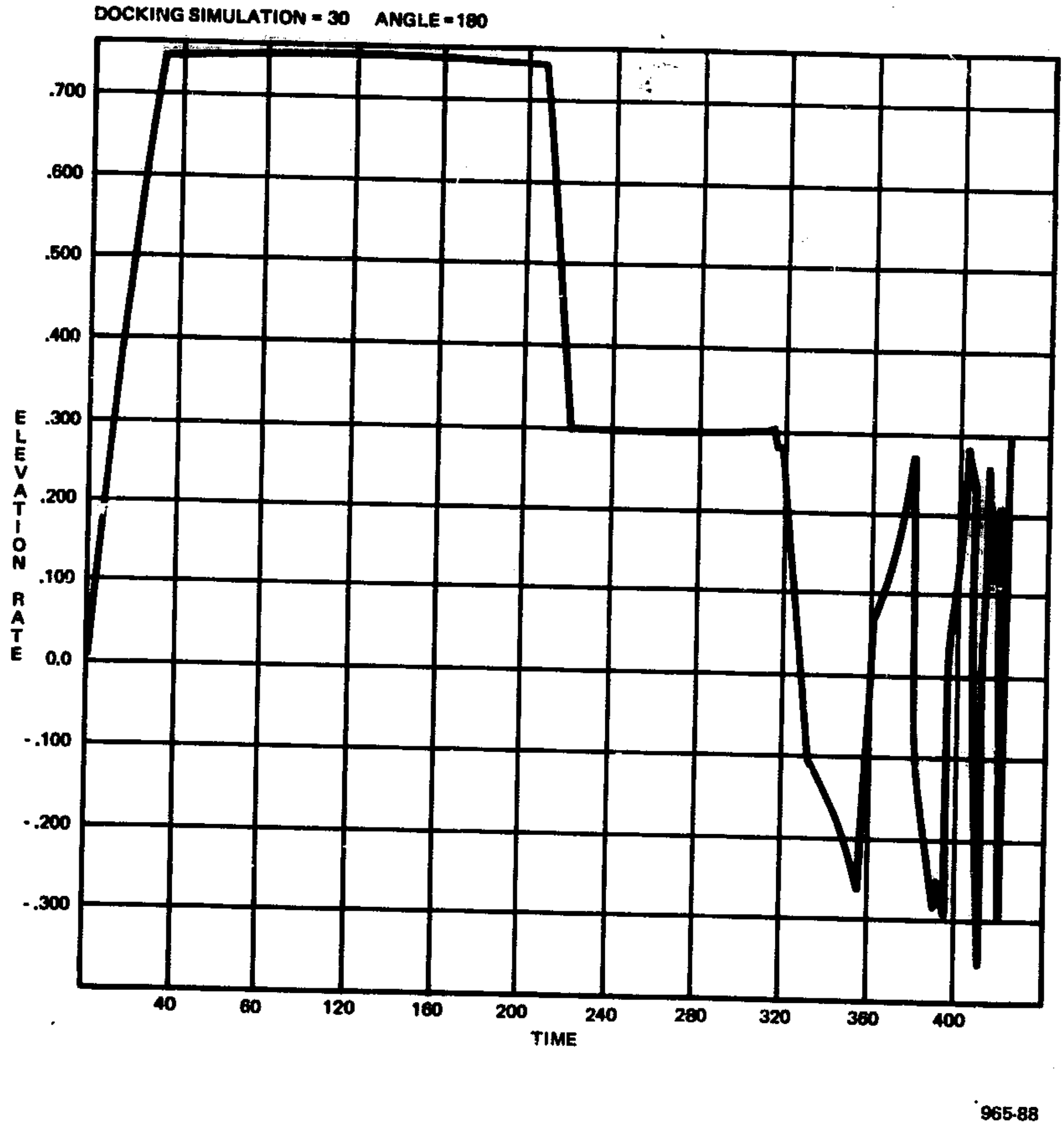

Figure 4-14. Elevation Rate VS. Time For Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

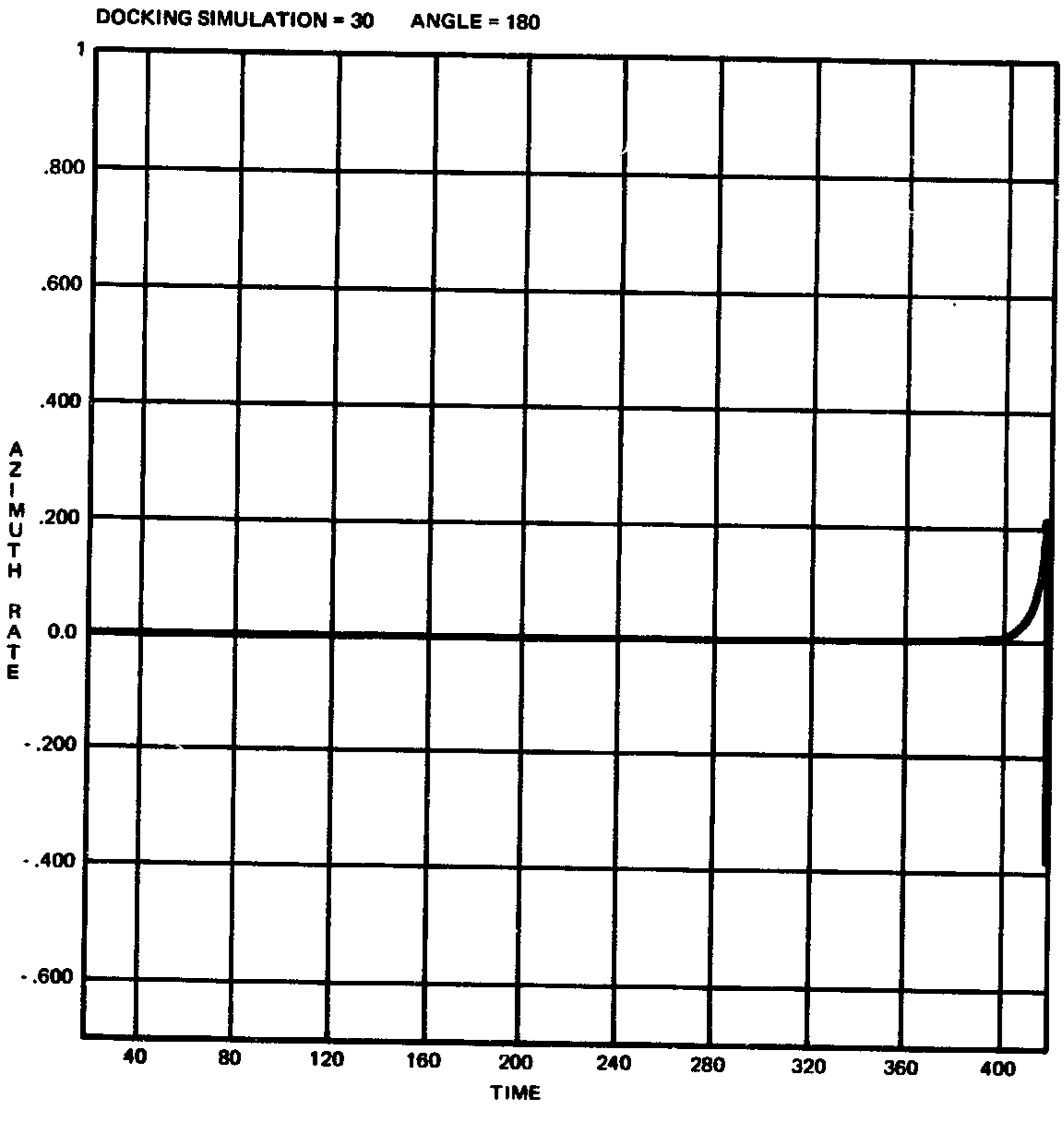

Figure 4-15. Azimuth Rate VS. Time for Intermediate Range = 1000 Meters and Go-Around Angle = 180 Deg.

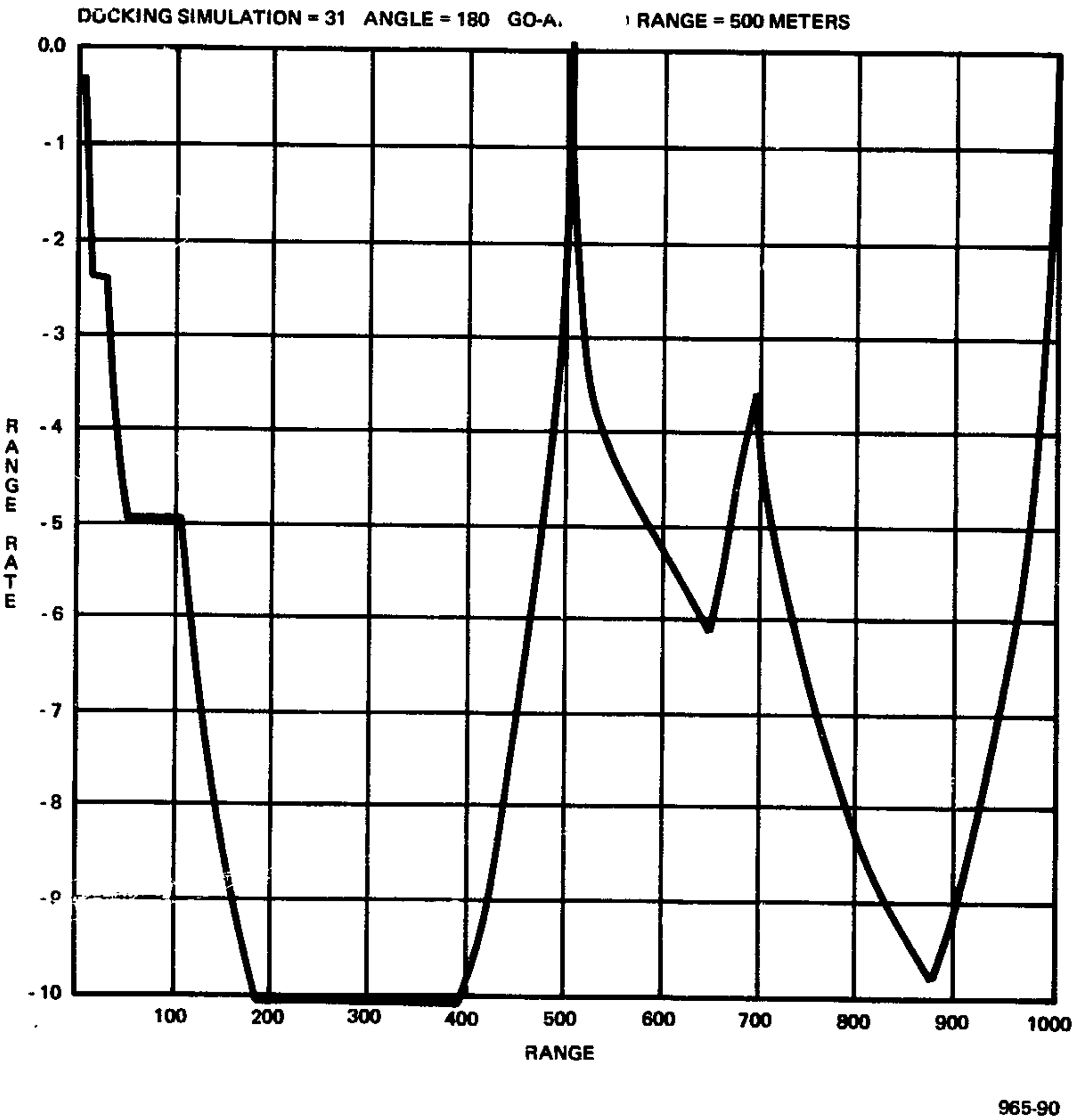

Figure 4-16. Range - Range Rate Phase Plane for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

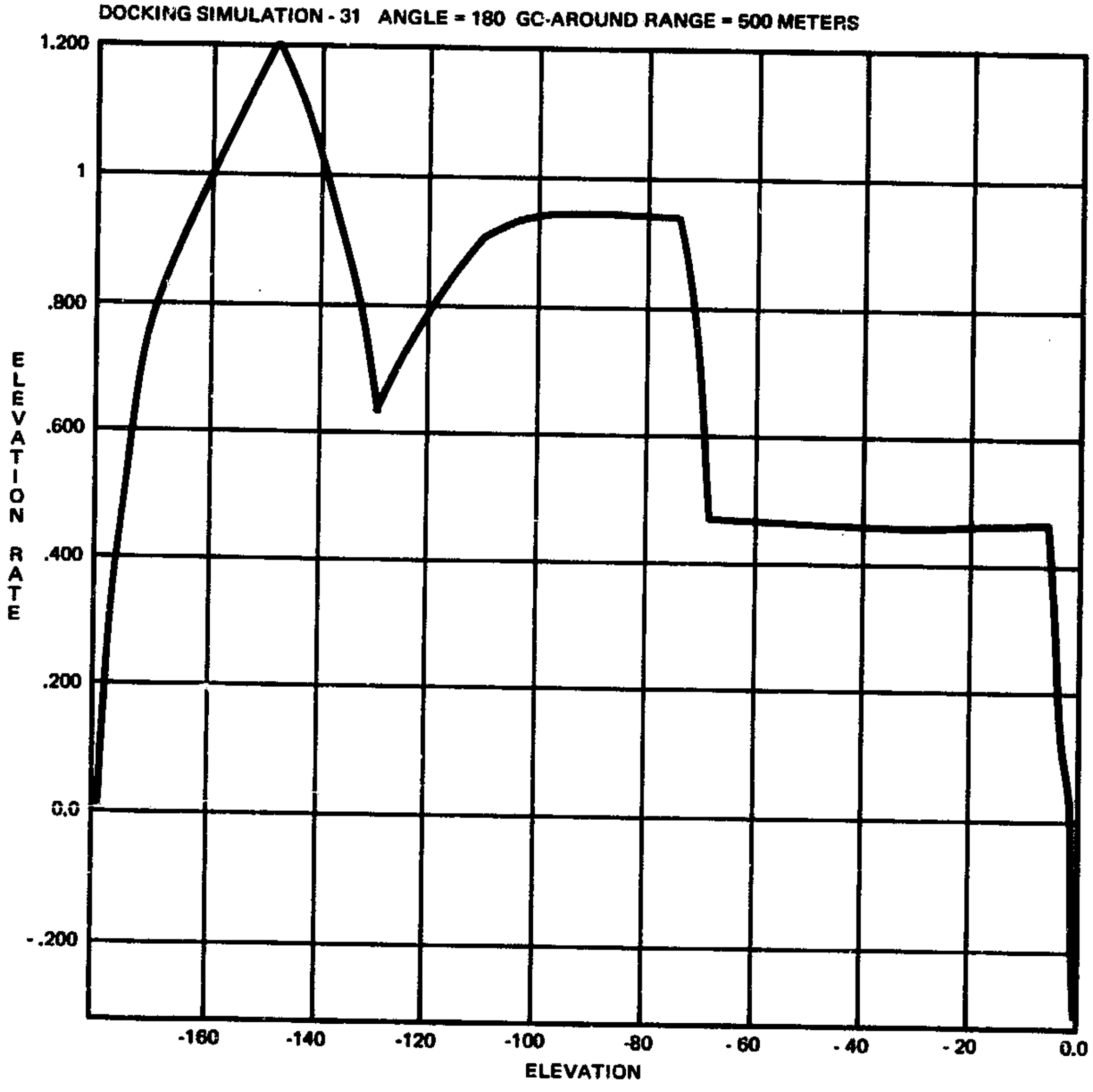

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Figure 4-17. Elevation - Elevation Rate Phase Plane for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

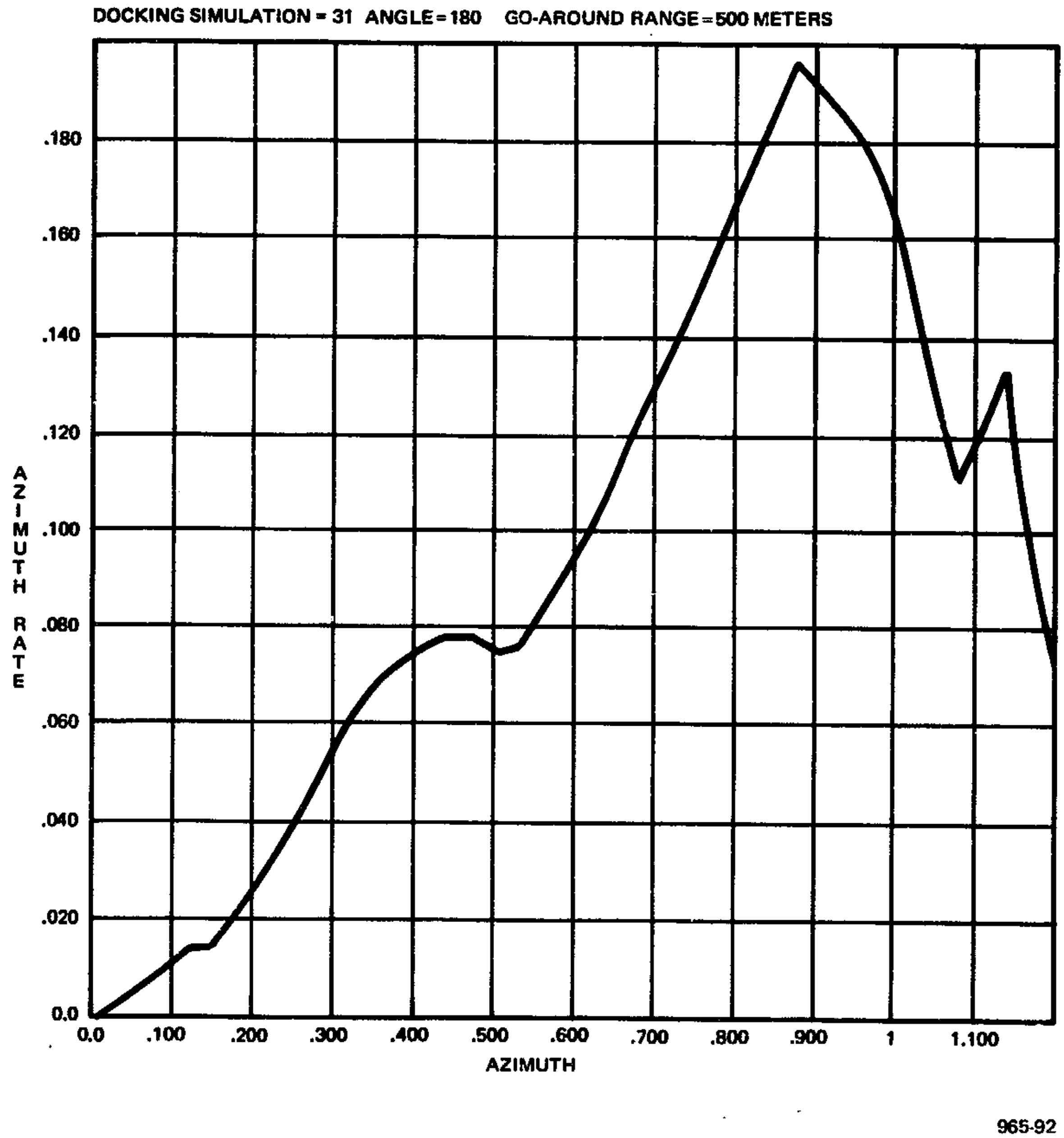

Figure 4-18. Azimuth - Azimuth Rate Phase Plane for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

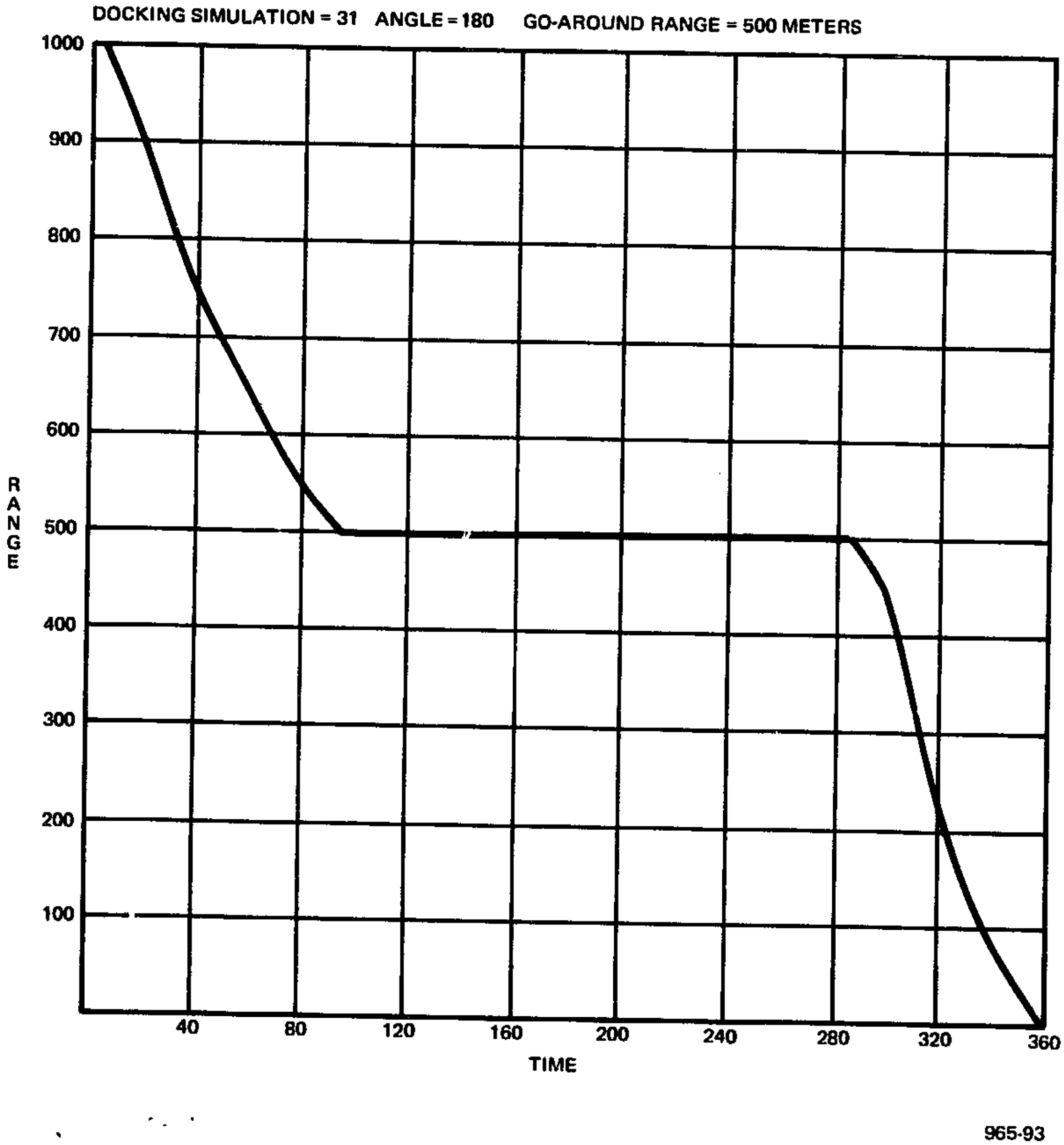

Figure 4-19. Range VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

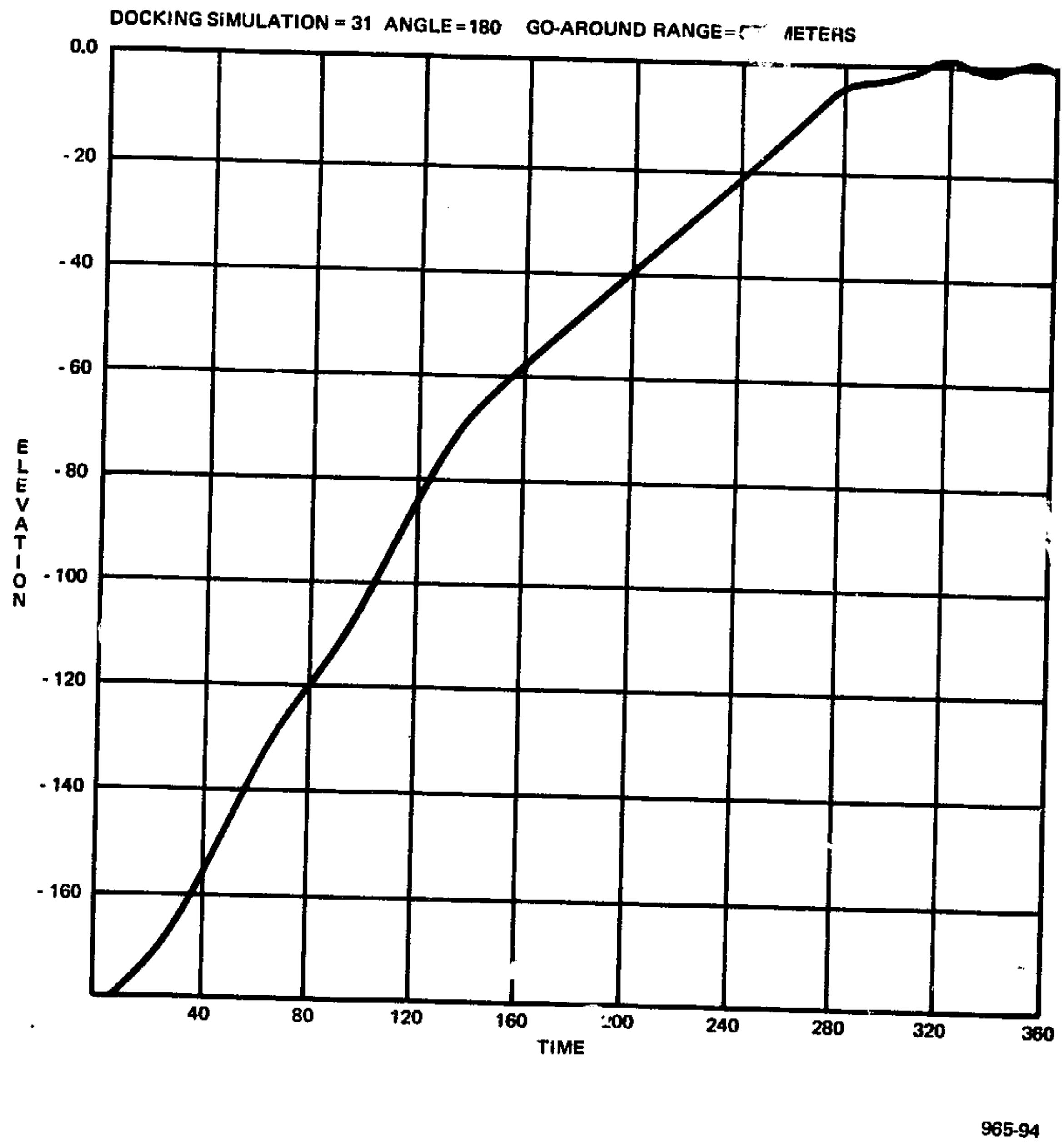

Figure 4-20. Elevation VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

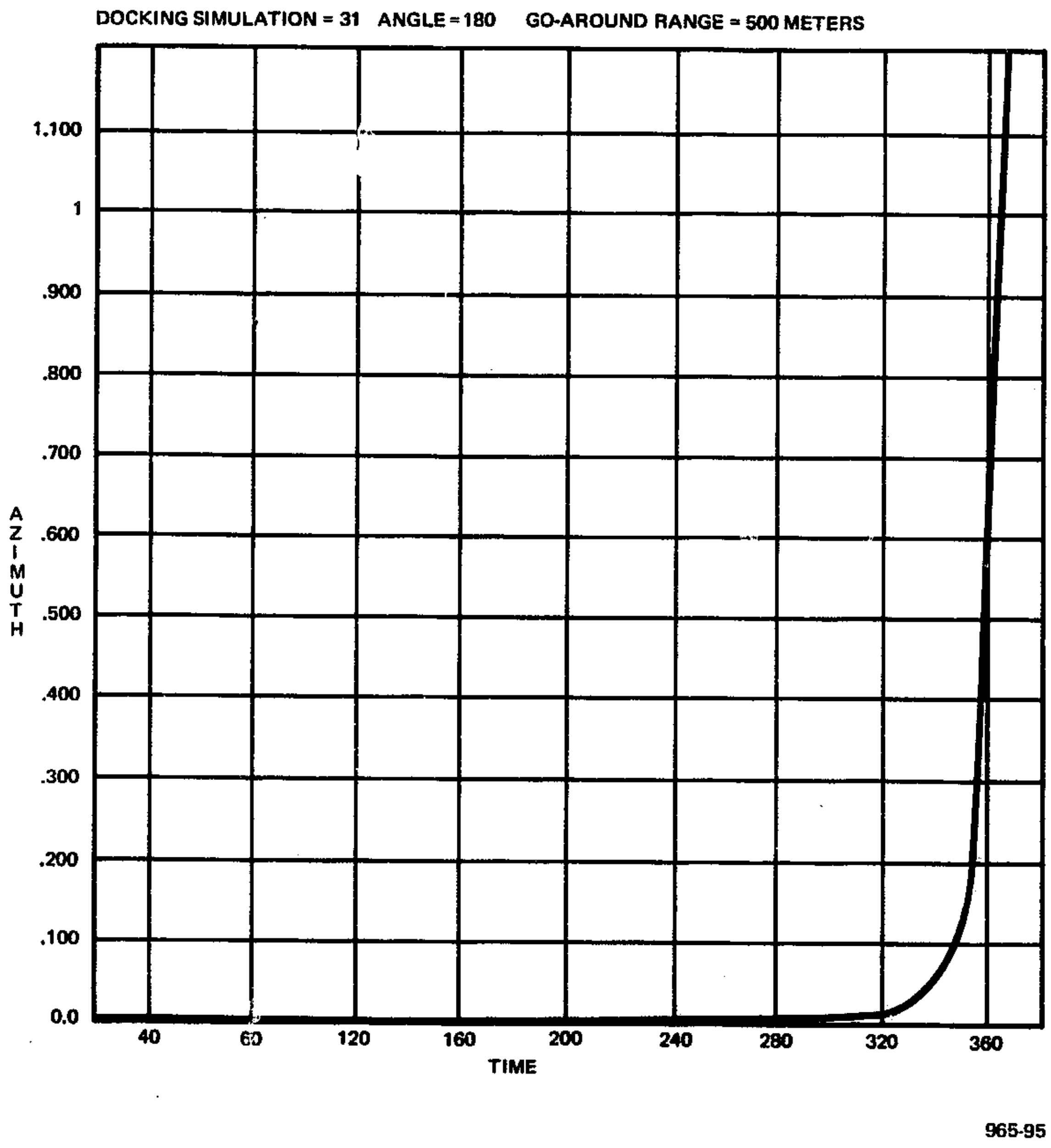

Figure 4-21. Azimuth VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

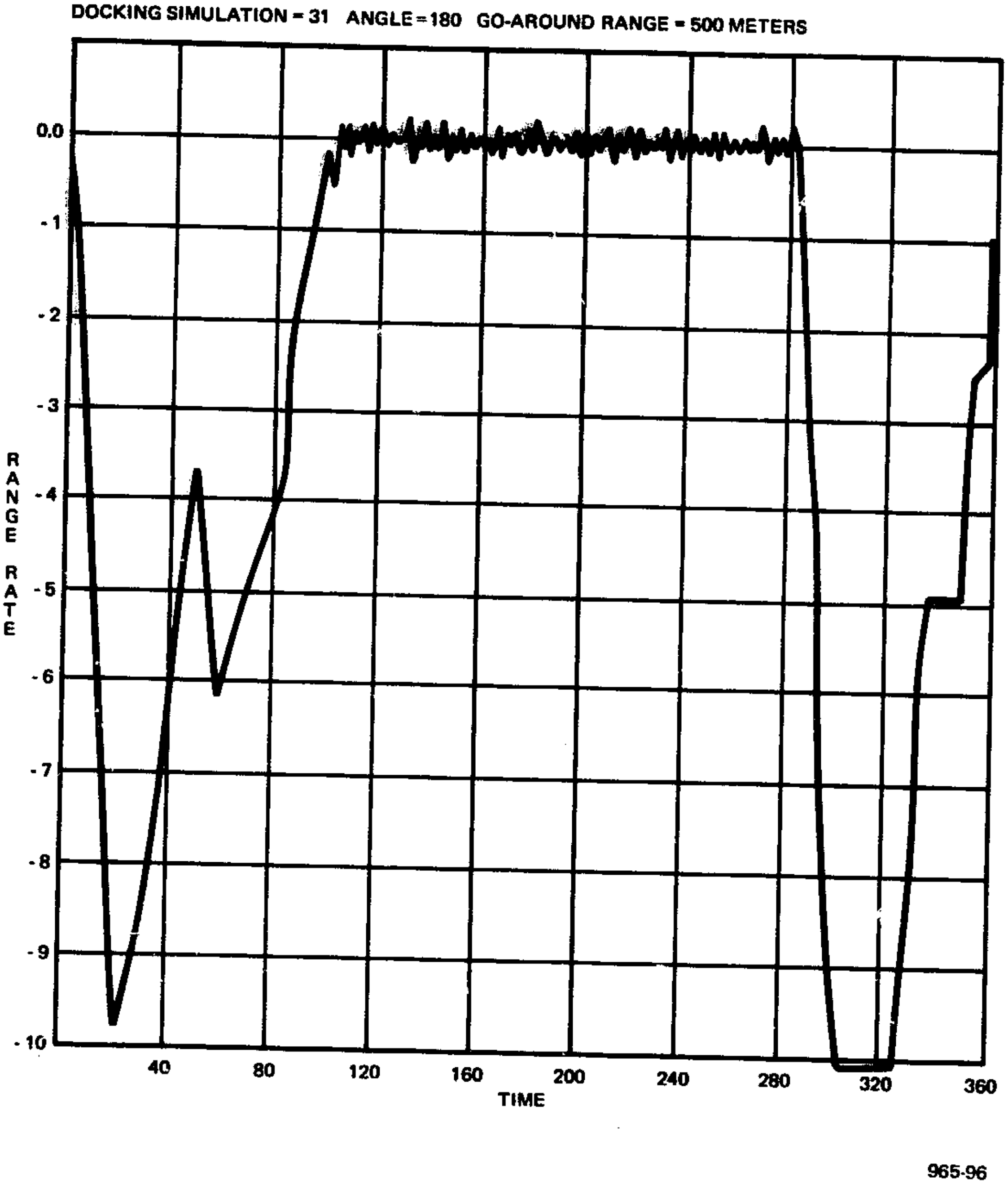

Figure 4-22. Range Rate VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

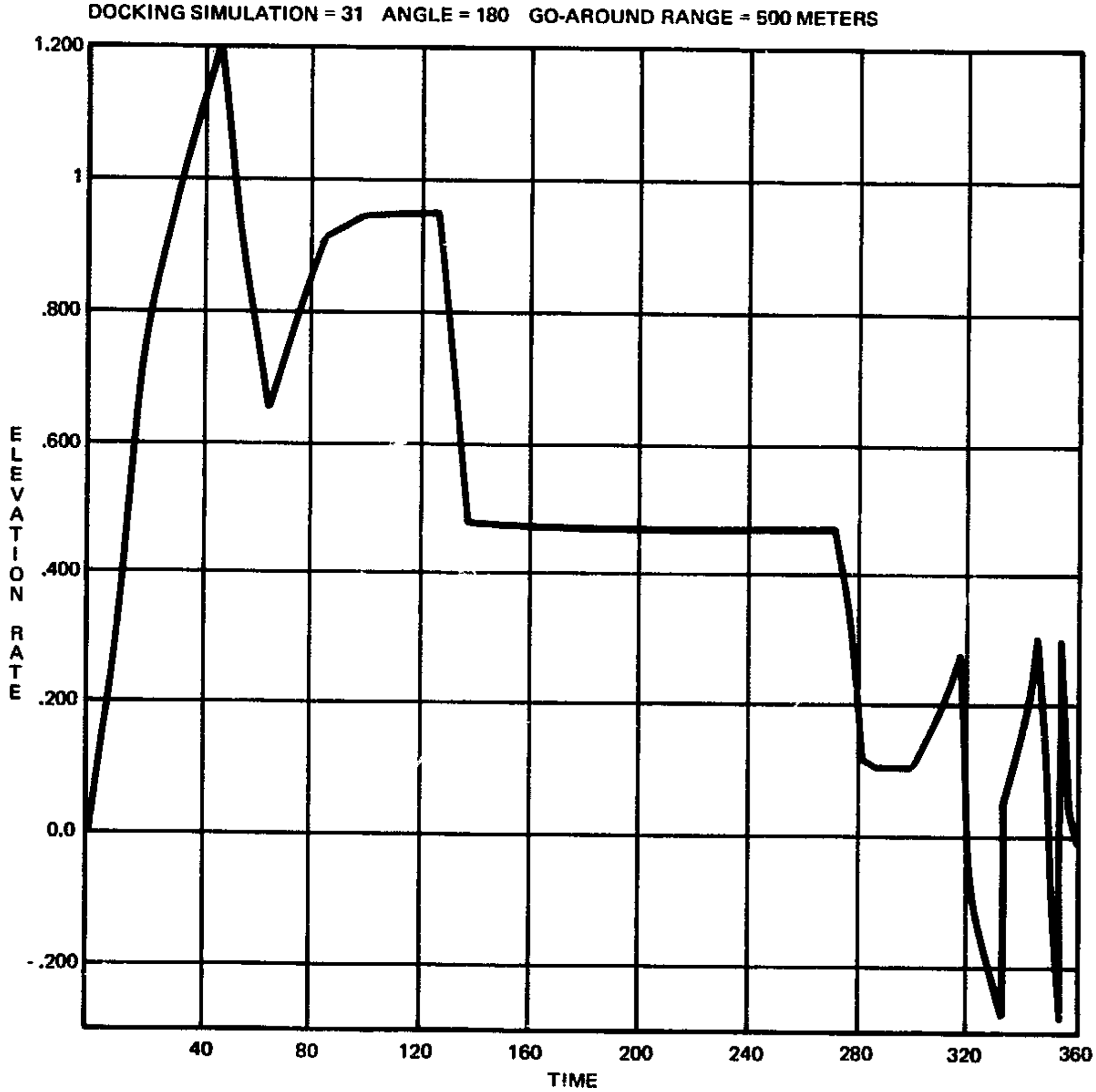

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Figure 4-23. Elevation Rate VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

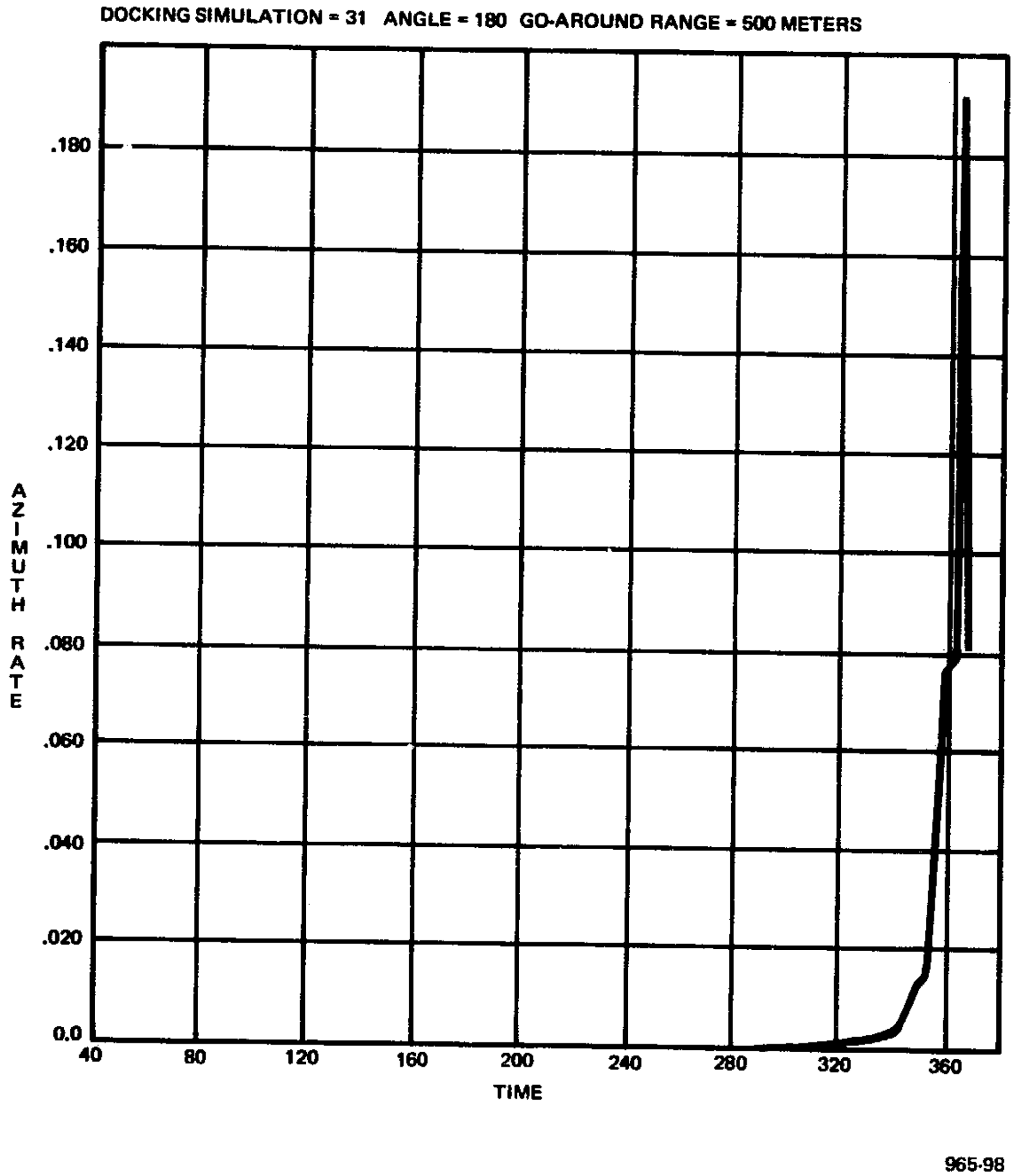

Figure 4-24. Azimuth Rate VS. Time for Intermediate Range = 500 Meters and Go-Around Angle = 180 Deg.

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APPENDIX F

SPACE TUG CONTROL
ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

The control function as used herein includes both attitude and translational control.

The purpose of attitude control is to maintain stability and to execute guidance maneuvers. Attitude control is accomplished by gimballing thrust vector control engines or firing reaction control system (RCS) engines. Translational control will also be used in executing guidance maneuvers. Translational control is achieved by throttling the main engine or firing the appropriate RCS thrusters. Main engine throttling capability will be particularly useful for the lunar landing mission. RCS translational control will be of particular value when used for docking maneuvers.

Major emphasis was placed on the synchronous orbit missions (SOM) for this study. However, hardware and computer software estimates for control functions are included for the tug missions which dictate design requirements for the astrionic system.

2.0 STUDY GUIDELINES AND GROUNDRULES

- Capability for manual and remote override of the automatic control function will be provided.
- Only one main engine per propulsion module.
- All control law functions will be performed in the digital computer.
- The guidance function shall provide attitude errors, attitude rate errors, position errors, and position rate errors to the control function as needed.
- During burn, the main engine will be used for pitch and yaw attitude control; the RCS will be used for roll attitude control.
- During coast, three-axis attitude control will be provided by the RCS.
- For the four stage Saturn V configuration, the Instrument Unit will be deleted and the astrionic module will assume complete control from ground launch.
- RCS engine control package and main engine actuators mounted on the PM are excluded from study at this time.

3.0 SUMMARY OF RESULTS

3.1 SOFTWARE REQUIREMENTS

The software requirements for the control function of the space tug missions are listed in Table 3-1. Note the following for proper interpretation of the tabulated values:

- Memory requirements include 25% contingencies.
- Time requirements include 15% contingencies.

Table 3-1. Control Function Software Requirements Summary for the Space Tug Missions

MISSION	FUNCTION	TOTAL (32-BIT) MEMORY	EQUIV. ADD'S	CALLS/SEC	EQUIV. ADD'S/SEC
SYNCHRONOUS ORBIT MISSION (EXPENDABLE)	* TVC RCS	1030 1044	673 1508	10 20	6730 30160
SYNCHRONOUS ORBIT MISSION (REUSABLE)	TVC RCS	1720 1044	673 1508	10 20	6730 30160
FOUR STAGE SATURN V (INCLUDES ALL STAGES)	TVC RCS	5215 1044	1298 1508	10 20	12980 30160
EARTH ORBITAL OPERATIONS	TVC RCS	1720 1044	673 1508	10 20	6730 30160
UNMANNED PLANETARY	TVC RCS	1720 1044	673 1508	10 20	6730 30160
REUSABLE NUCLEAR SHUTTLE	TVC RCS	1720 1044	673 1508	10 20	6730 30160
LUNAR ORBITAL OPERATIONS	TVC RCS	1720 1044	673 1508	10 20	6730 30160
LUNAR LANDING	TVC RCS	2020 1310	1350 1800	10 20	13500 36000

* THRUST VECTOR CONTROL (TVC)

- The RCS software estimates include the combined impact of 3-axis attitude and 3-axis translational control.
- Software requirements for RCS attitude control were assumed to be identical to RCS translational control.
- The lunar landing mission software requirements include engine throttling software estimates. A throttleable engine requirement has been identified only for the lunar landing mission.

3.2 HARDWARE REQUIREMENTS

With the exception of standard interface units, additional hardware may be required only for the four stage Saturn V configuration to enhance vehicle stability and load relief during earth launch. This would include the following:

(1) One 3-axis rate gyro package

(2) One 2-axis lateral accelerometer package

The requirements for the above hardware will be established in follow-on study effort.

3.3 MODULE INTERFACES

Control interfaces with the propulsion module, the electrical support equipment, and the command function for the synchronous orbit mission are illustrated in Figure 3-1.

4.0 DETAILED ANALYSIS (TRADE STUDIES)

4.1 SYNCHRONOUS ORBIT MISSIONS CONTROL ANALYSIS

4.1.1 Unmanned-Expendable Tug Stage

4.1.1.1 Hardware Requirements

The control hardware requirements are minimal. Assuming a data bus in used, only interface units to and from the data bus will be required (see Figure 4-1). With a centralized I/O, no additional hardware will be required.

The space tug synchronous orbit mission TVC subsystem control law will require vehicle body-referenced attitude error inputs only. These will be transmitted to the control function by the guidance function. The guidance function utilizes navigation sensor inputs to compute the attitude error signals.

RCS subsystem control law inputs required will be vehicle body-referenced attitude errors and attitude rate errors. The attitude error signal will again be transmitted to the control subsystem by the guidance subsystem. Attitude rate errors will be determined by a combination of guidance and control function computations.

Vehicle angular rate sensors will not be a hardware requirement because the required angular rate signals will be derived from IMU attitude inputs to the computer. If IMU sensor noise prohibits explicit rate derivation, pre-filtering and/or state estimation techniques can be employed. Associated software requirements for this application should be minimal. Rate derivation techniques have been proven on space vehicles such as Minuteman, LEM, and CSM.

4.1.1.2 Attitude Control Analysis

4.1.1.2.1 Thrust Vector Control (TVC) Subsystem. Pitch and yaw attitude control will be accomplished by TVC during the burn portions of the synchronous mission. The conceptual TVC subsystem is shown in Figure 4-2. The guidance and navigation measurements and computations will supply attitude error inputs to the control law. They will be multiplied by the control gain factors and then will be transmitted to the digital compensators. The compensator outputs will be the engine position commands. Engine position feedback, either electrical or mechanical, with shaping will serve to provide desired actuator responses to positioning command inputs. With mechanical feedback the servo position shaping will be inherent to the actuator hardware.

Effective damping will be provided by the filter designs (lead-lag filters). This eliminates the requirement for explicit rate feedback. The time-varying control gains will be adjusted to anticipated variations in thrust, vehicle center of gravity, and vehicle moment of inertia in order to maintain suitable stability margins and a near constant control natural frequency.

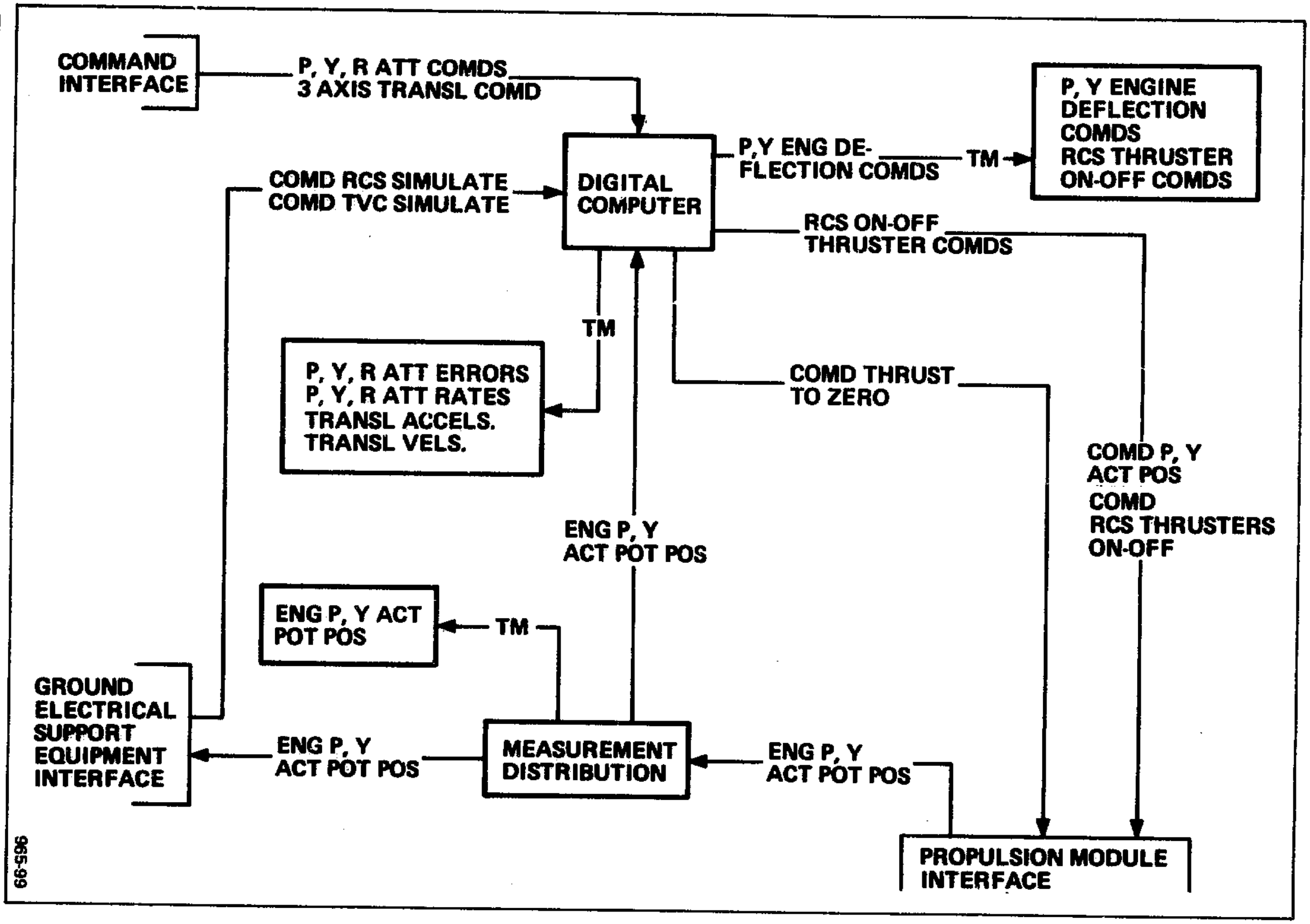

Figure 3-1. Space Tug Control Functional Interconnect—Synchronous Orbit Mission Interfaces

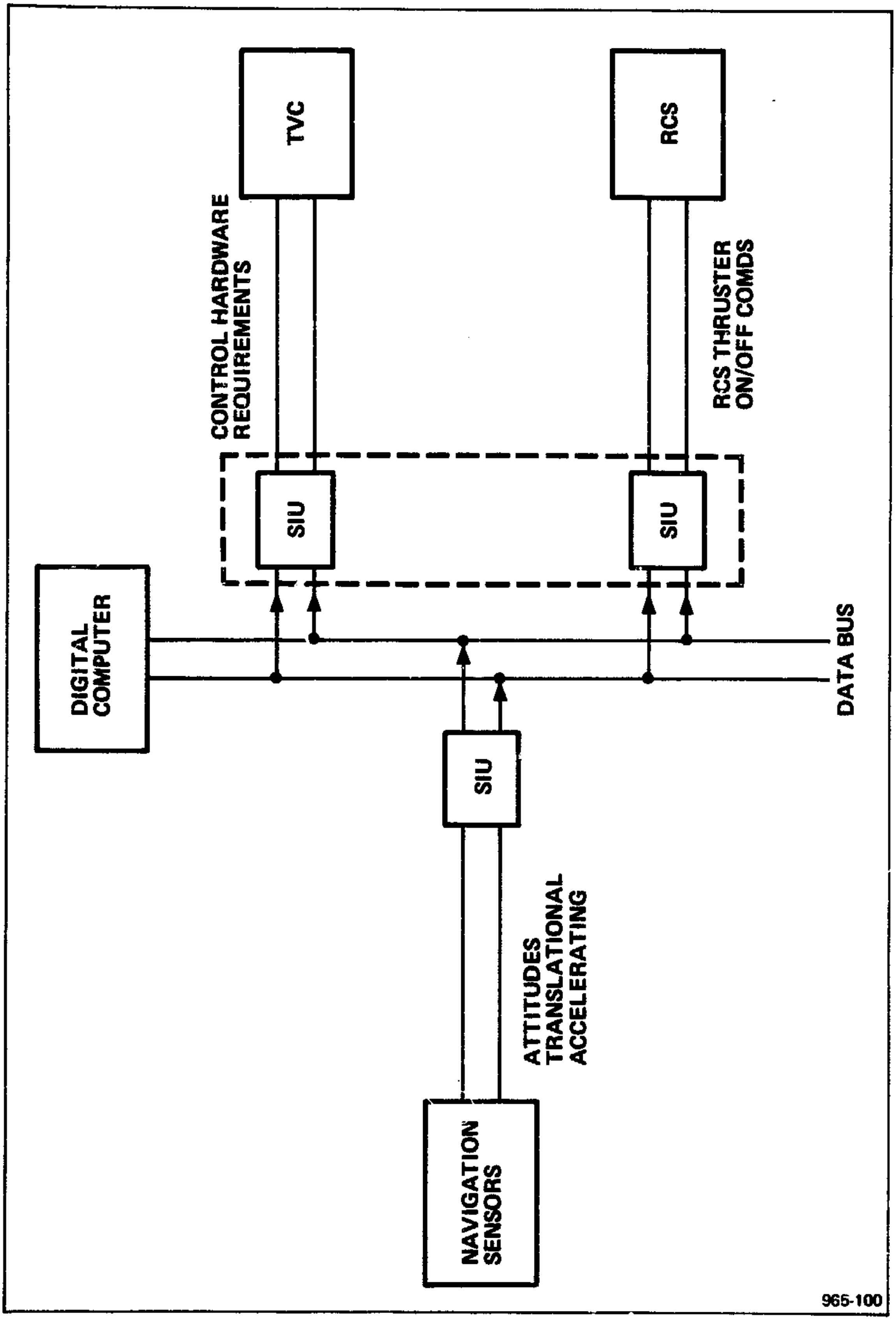

Figure 4-1. Synchronous Orbit Missions Control Hardware Requirements and Interfaces With Data Bus Integration

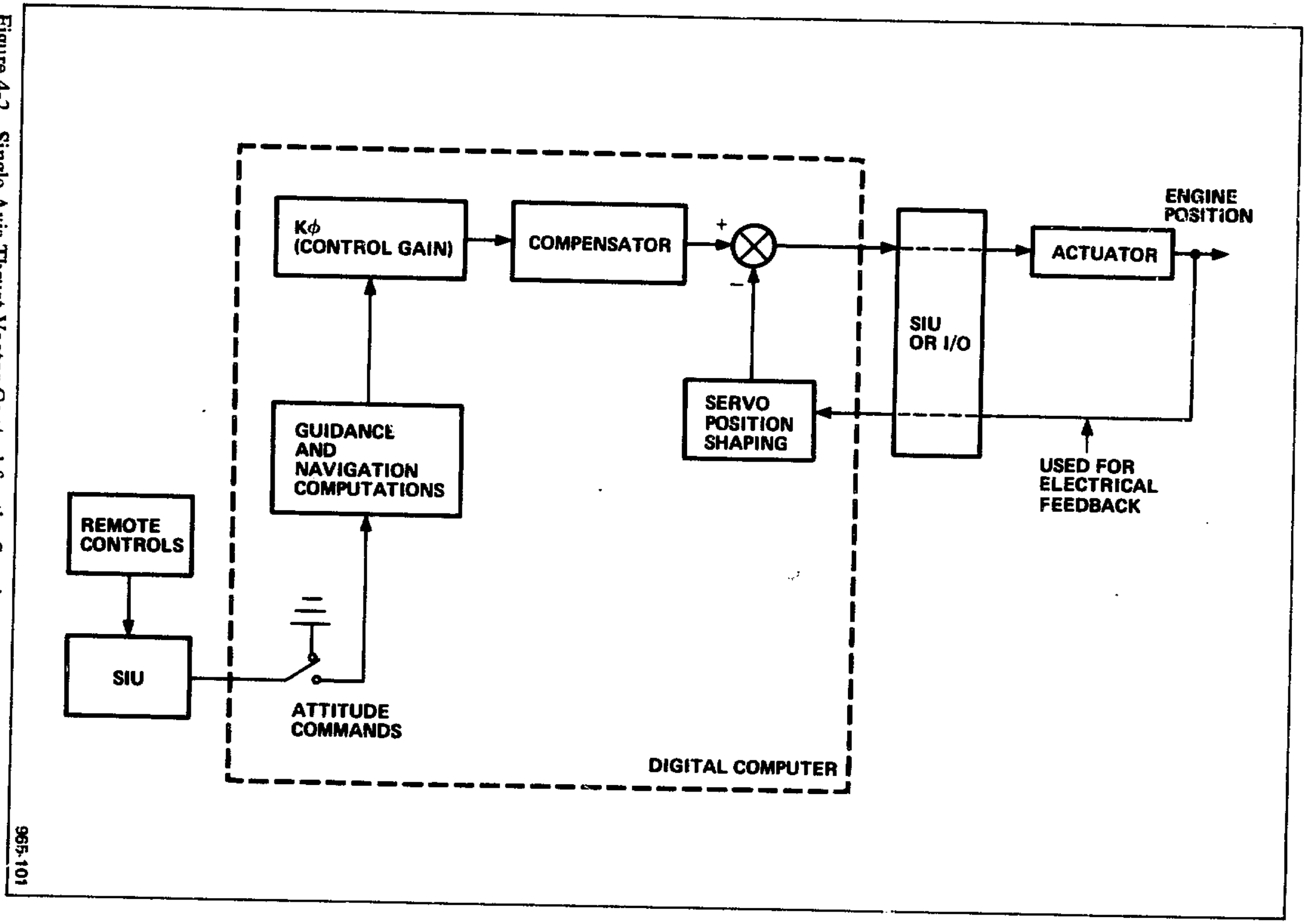

Figure 4-2. Single Axis Thrust Vector Control for the Synchronous Orbit Missions

TVC engine gimbal requirements which must be defined as part of the actuator design include the following:

- Maximum angular gimbal rate
- Maximum gimbal angle
- Bandpass frequency range of the actuator control

Simulation tools are most effective for determining the appropriate TVC engine gimbal requirements based on maneuvers and potential disturbances. Further definition of operating modes is required. Simulations should be exercised in follow-on work to establish appropriate gimbal requirements for the tug engine.

The TVC subsystem control law can be written as follows:

$$B_C = K_\phi \, D(Z) \Delta_\phi \quad \text{(pitch or yaw)}$$

where

- B_C is the engine deflection command signal
- K_ϕ is the attitude control gain
- $D(Z)$ is the digital filter
- Δ_ϕ is the attitude error signal before filtering

This equation will be updated 10 times per second during a burn.

The pitch and yaw filters are assumed to be eighth-order and of the form:

$$D(Z) = \frac{A_0 + A_1 Z^{-1} + A_2 Z^{-2} + \cdots + A_8 Z^{-8}}{1.0 + B_1 Z^{-1} + B_2 Z^{-2} + \cdots + B_8 X^{-8}}$$

Multiple-case filters are not considered for the unmanned expendable stage of the synchronous orbit mission.

The control gains will be time varying for computer implementation purposes. The desired values of the control gains for a particular time of flight will have previously been determined by incorporating predicted vehicle data in control design. The implementation equations will be based on straight line approximations to the gain versus time profiles. The values of K_ϕ for each straight line segment beginning at time T_I and ending at time T_F will be incremented according to the following equation:

$$\Delta K_\phi = \frac{K_{\phi F} - K_{\phi I}}{(T_F - T_I)\ 10} \quad \text{(computed five times during each burn)}$$

It will be assumed that five straight line segments are required for each burn. The recursive equation for K_ϕ is then

$$K_{\phi i} = K_{\phi i-1} + \Delta K_\phi \qquad \text{(computed 10 times per second)}$$

Actuator servo loop will require either digital or mechanical feedback compensation. The configuration of Figure 4-2, assuming fourth-order servo position shaping filters, will be considered for computer sizing.

A summary of the groundrules used to assess the computer requirements for the TVC subsystem follows:

- The control law will be computed 10 times per second.
- The digital filter implementation will be cascade-direct.
- The memory requirements will be sized for 32-bit words.
- The computer speed requirements will be based on equivalent ADD's per second.
- A MULTIPLY execution will be assumed equal to four equivalent ADD's.
- Each non-arithmetic operation will be equal to one equivalent ADD.
- Redundancy considerations will not be included in the results.

The unmanned expendable stage TVC computer requirements are summarized in Table 4-1. Note that the filter computation computer requirements are most significant for TVC.

Table 4-1. TVC Computer Requirements for the Unmanned-Expandable Synchronous Orbit Mission

	NUMBER REQUIRED	MULT.	ADD/SUB	NON-ARITH	DATA	TOTAL MEMORY	EQUIV. ADDS	CALLS/SEC	EQUIV. ADDS/SEC
8/8 FILTER (REFERENCE F-11)	2	17	17	64	38	272	298	10	2,980
4/4 FILTER (REFERENCE F-12)	2	9	9	40	25	166	170	10	4,700
ΔK_ϕ	20	2	2	4	6	280	—	—	—
K_ϕ	2	—	1	4	3	16	10	10	100
INPUT/OUTPUT	6	—		3	1	24	18	10	180
LIMITERS	2	—	2	16	10	56	36	10	360
MISC. OPERATIONS	—	—	2	4	4	10	6	10	60
TOTAL	—	92	100	350	282	824	538	10	15,380

4.1.1.2.2 Reaction Control Subsystem. RCS phase plane logic design is generally based on two major considerations:

(1) Minimizing RCS fuel requirements

(2) Minimizing the time required to drive the vehicle attitude error and rate error ($\Delta\phi, \Delta\dot{\phi}$, respectively) to the limit cycle

A trade relationship exists between (1) and (2). Therefore, the phase plane design for a particular mission phase depends heavily on the mission phase requirements. For example, fast response time is required during a lunar landing mission. The astronauts may need to maneuver quickly in order to avoid a boulder, etc. Simultaneously, RCS fuel minimization is a requirement. These factors are weighted accordingly and then incorporated into the phase plane design.

Parabolic switching lines are often used when time is of essence. Straight lines are often used when time is not critical and simplicity is desired. The illustrations shown in Figure 4-3 are examples of such phase plane portraits. Straight line phase plane portraits will likely be adequate for the synchronous mission.

The RCS attitude control implementation (Figure 4-4) will be used for roll axis control during burn modes and for three-axis attitude control during coast modes. The groundrules and assumptions used for assessment of the RCS requirements are as follows:

- The three axes will be uncoupled for nominal control.
- Only the nominal control case will be considered.
- The phase plane control law inputs will be attitude errors and attitude rate errors.
- The computational solution rate will be 20 samples/sec (Reference F-15).
- A coarse deadband of 5.0 deg will be selected for coasting (References F-1 and F-7).
- A fine deadband of 3.0 deg will be selected for docking purposes (References F-3, F-16, and F-17).

The phase plane portrait of Figure 4-3 (b) could conceivably be used for the expendable stage attitude control during docking maneuvers. This phase plane portrait would satisfy typical attitude control docking requirements of (1) maximum alignment angle of 4 degrees and (2) maximum angular rate of 0.5 deg/sec. This is possible because the attitude error deadband (3°) and the attitude rate error deadband ($.2^\circ$/sec) are less than these maximums. The control equation flow for this RCS logic is shown in Figure 4-5 (see Reference F-15). The three-axis attitude control function computer requirements, based on the equation flow diagram, are summarized in Table 4-2.

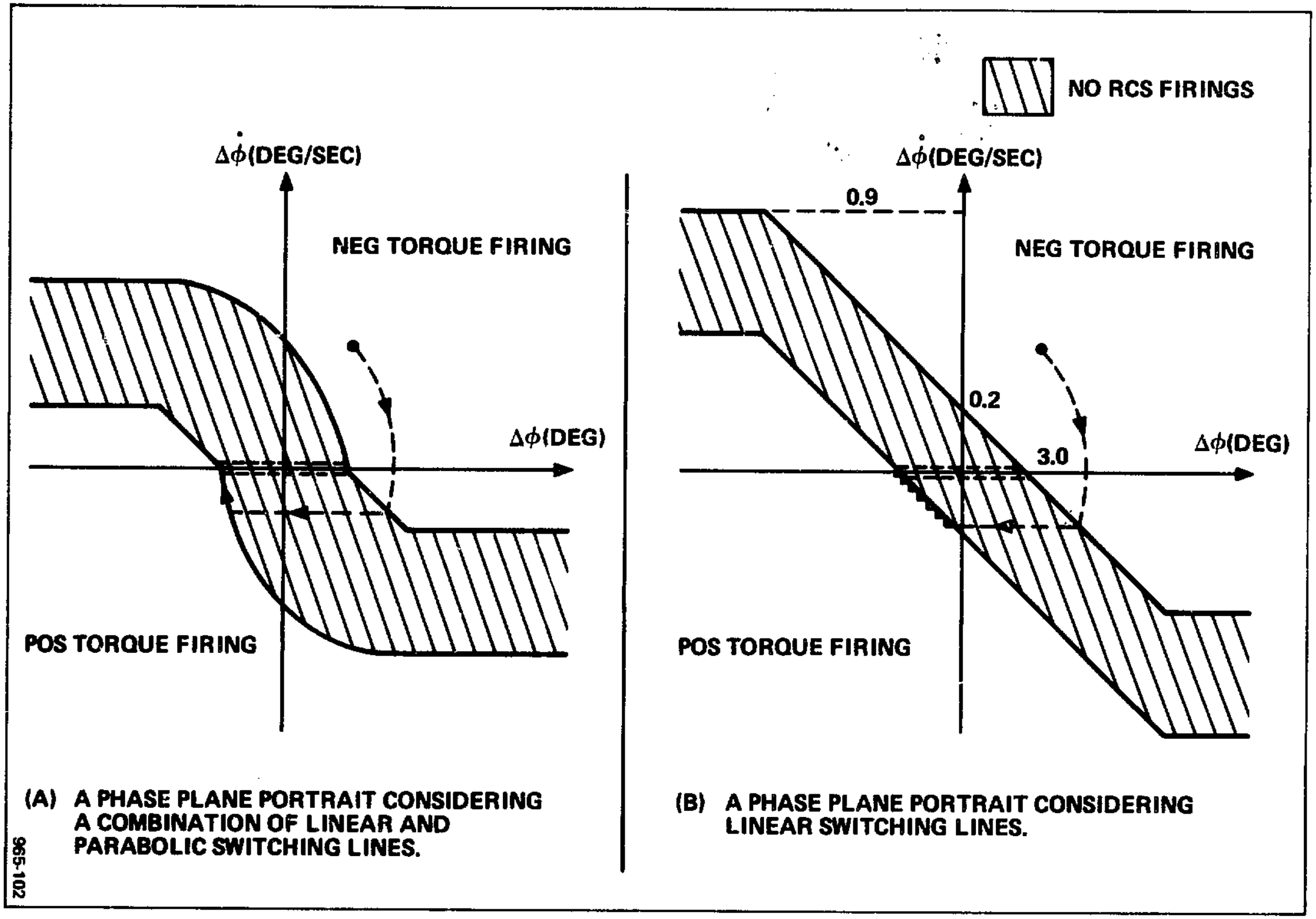

Figure 4-3. RCS Phase Plane Portraits

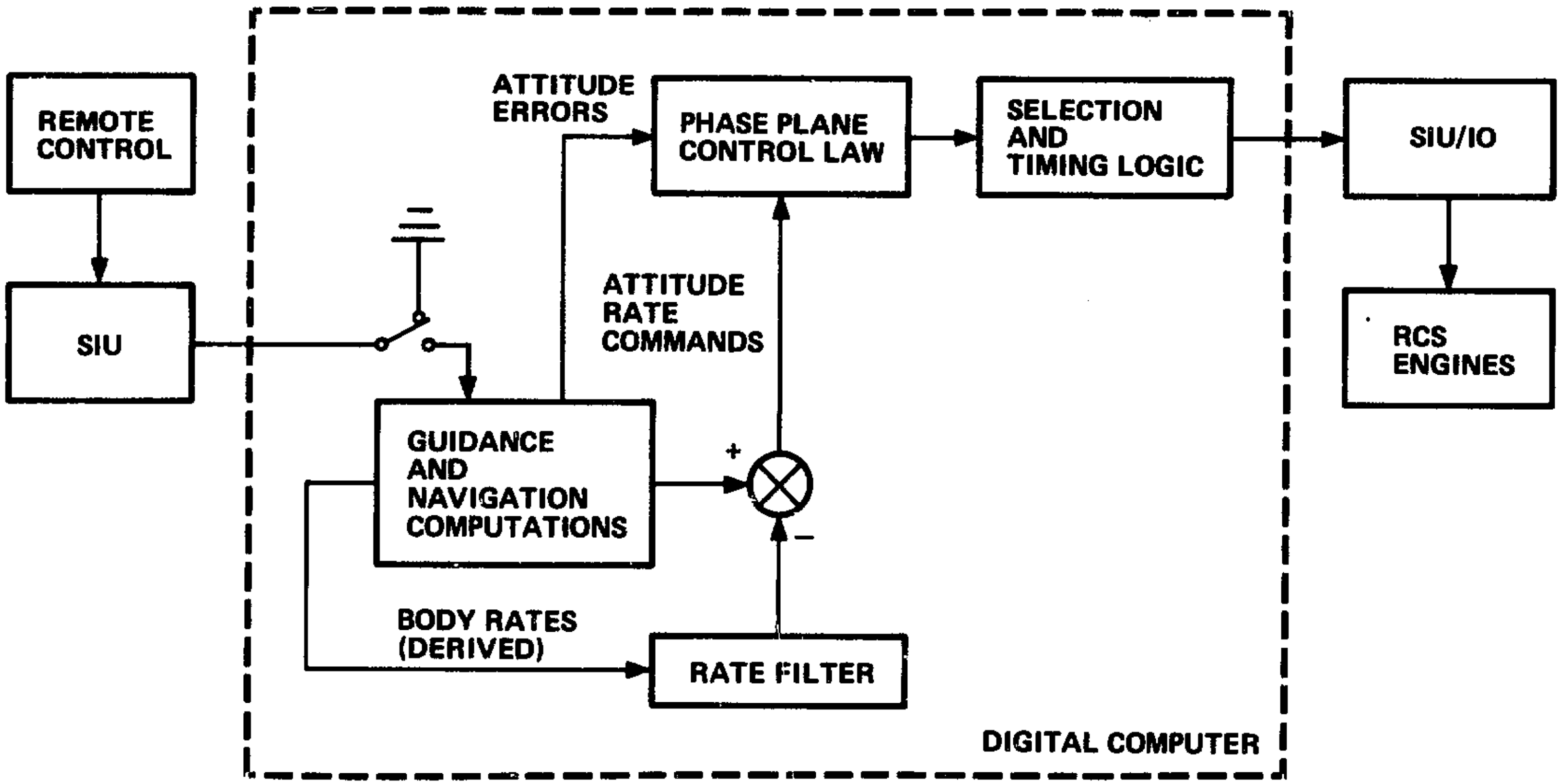

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Figure 4-4. RCS Attitude Control

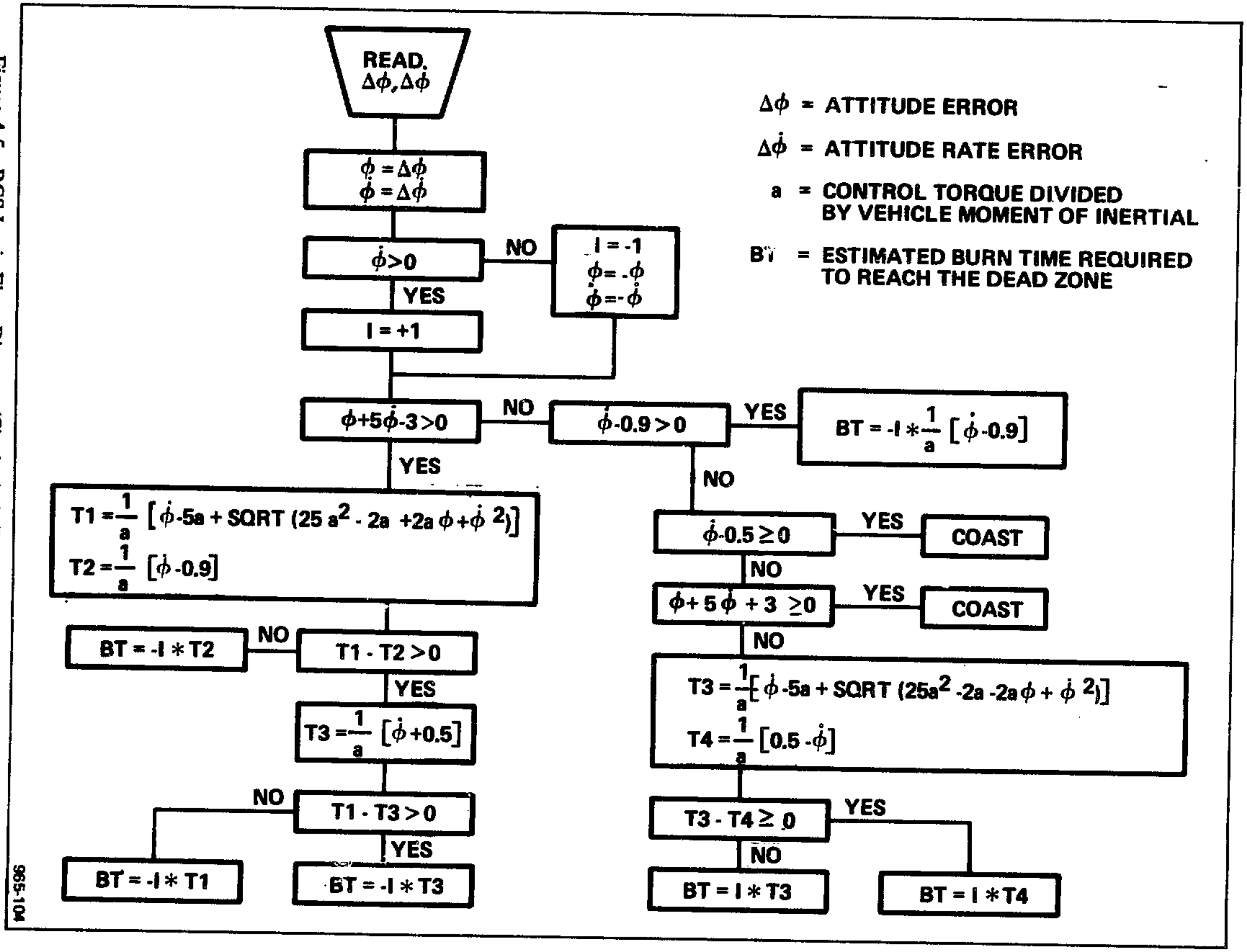


Figure 4-5. RCS Logic Flow Diagram (Single Axis Equation)

Table 4-2.

MEMORY REQUIRED	EQUIV. ADD'S	CALLS/SEC	EQUIV. ADD'S/SEC
417	603	20	12,060

4.1.1.3 Translational Control (TC) Analysis

Three axis translational control will be provided. Translation along a particular axis will be provided by firing the appropriate RCS thrusters. TC implementation will be analogous to the RCS implementation in the following respects:

- The range, elevation, and azimuth axes are analogous to the pitch, yaw and roll axes.
- Translational rate error and position error signals are analogous to the attitude rate error and attitude error respectively.
- Phase plane logic will again be employed.
- The optimal time phase plane trajectory will be a parabolic function.

TC will be designed such that the following docking requirements will be met:

(1) The maximum lateral offset will be 1 ft.

(2) The maximum lateral rate will be 0.5 ft/sec.

(3) The maximum closing velocity will be 1.0 ft/sec (see Reference F-3).

TC software requirements will be assumed identical to those of the previous section.

4.1.2 Unmanned-Reusable Stage

4.1.2.1 Hardware Requirements

These are the same as those described in Section 4.1.1.1.

4.1.2.2 Attitude Control Analysis

4.1.2.2.1 Thrust Vector Control Subsystem. The TVC subsystem is generally the same as that described in Section 4.1.1.2.1. However, because of vehicle configuration changes, it will be necessary to switch filters at least once per tug stage during the mission. The impact will be software storage requirements. The configurations and filter requirements are summarized in Table 4-3.

Table 4-3.

	CONFIGURATION *		FILTER REQUIREMENTS
TUG 1	BURN NUMBER 1 BURN NUMBER 2 BURN NUMBER 3	TUG 1 + TUG 2 + PAYLOAD TUG 1 ONLY TUG 1 ONLY	USE FILTER A ONLY USE FILTER B ONLY USE FILTER B AGAIN
TUG 2	BURN NUMBER 1 BURN NUMBER 2 BURN NUMBER 3 BURN NUMBER 4	TUG 2 + PAYLOAD TUG 2 + PAYLOAD TUG 2 ONLY TUG 2 ONLY	USE FILTER C ONLY USE FILTER C AGAIN USE FILTER D ONLY USE FILTER D AGAIN
* REFERENCE APPENDIX A, REUSABLE SYNC ORBIT MISSION FOR MISSION PROFILE.			

The computer TVC memory requirements will increase over that required for the unmanned-expendable stage mission as shown below:

	Memory Locations Required
Unmanned-expendable stage mission	824
Due to additional filter requirements	272
Due to additional ΔK_ϕ requirements*	280
Total for unmanned reusable stage mission	1,376

4.1.2.2.2 Reaction Control Subsystem. The RCS implementation will be the same as that described in Section 4.1.1.2.2.

4.1.2.3 Translational Control (TC)

TC will be the same as that described in Section 4.1.1.3.

4.2 GENERAL CONTROL CONSIDERATION

4.2.1 Thrust Vector Control Analysis for Other Space Tug Missions

4.2.1.1 Hardware Requirements

The control function for all tug missions, except the four stage Saturn V, have the same hardware requirements. Additional control sensors may be required for the four stage Saturn V mission. These are (1) a three-axis rate gyro package and (2) a two-axis lateral control accelerometer package. For all other missions and mission phases, control inputs will

* The ΔK_ϕ storage requirements are based on four burns (20 straight line segments per axis).

be obtained from guidance and navigation functions (refer to Section 4.1.1.1). Other hardware requirements pertain to interface units to and from the data bus. The seven missions' hardware requirements are summarized pictorially in Figure 4-6. Physical and electrical characteristics of the candidate rate gyro and control accelerometer packages are listed in Table 4-4.

4.2.1.2 Attitude Control Law Comparison

4.2.1.2.1 Candidate Thrust Vector Control Laws. The following TVC laws applicable to space tug missions are briefly discussed in this section:

(1) The LM TVC laws

(2) The CSM TVC laws

(3) The Saturn IB TVC laws

(4) A modified TVC disturbance acceleration nulling scheme

The intent of these discussions is to present some of the present-day TVC laws and philosophies and their applicability for the space tug TVC subsystem.

(1) The LM TVC subsystem (Reference F-8) uses two basic modes of operation:

- The Acceleration Nulling Mode utilizes RCS for attitude control and TVC for disturbance angular acceleration nulling.
- The Attitude Control Mode employs TVC for pitch and yaw attitude control and RCS for roll control.

Because the LM TVC gimbal rate capability is limited (max gimbal rate of 0.2^o/sec), the Attitude Control Mode was designed for flight situations which impose only mild requirements on the attitude control subsystem.

Table 4-4.

SENSOR	INPUT POWER (WATTS)	INPUT RANGE (^{o}/SEC)	OPERATING TEMP. (^{o}C)	DIMENSIONS (HXWXL IN.)	WEIGHT (LBS)
RATE GYRO PACKAGE (REFERENCE F-13)	45	±20	-40 TO +74	3.62X7.06X 7.06	11.0
LATERAL ACCELER-OMETER PACKAGE (REFERENCE F-14)	8.4	±10	-73 TO +71	3.62X4.0X5.0	4.0

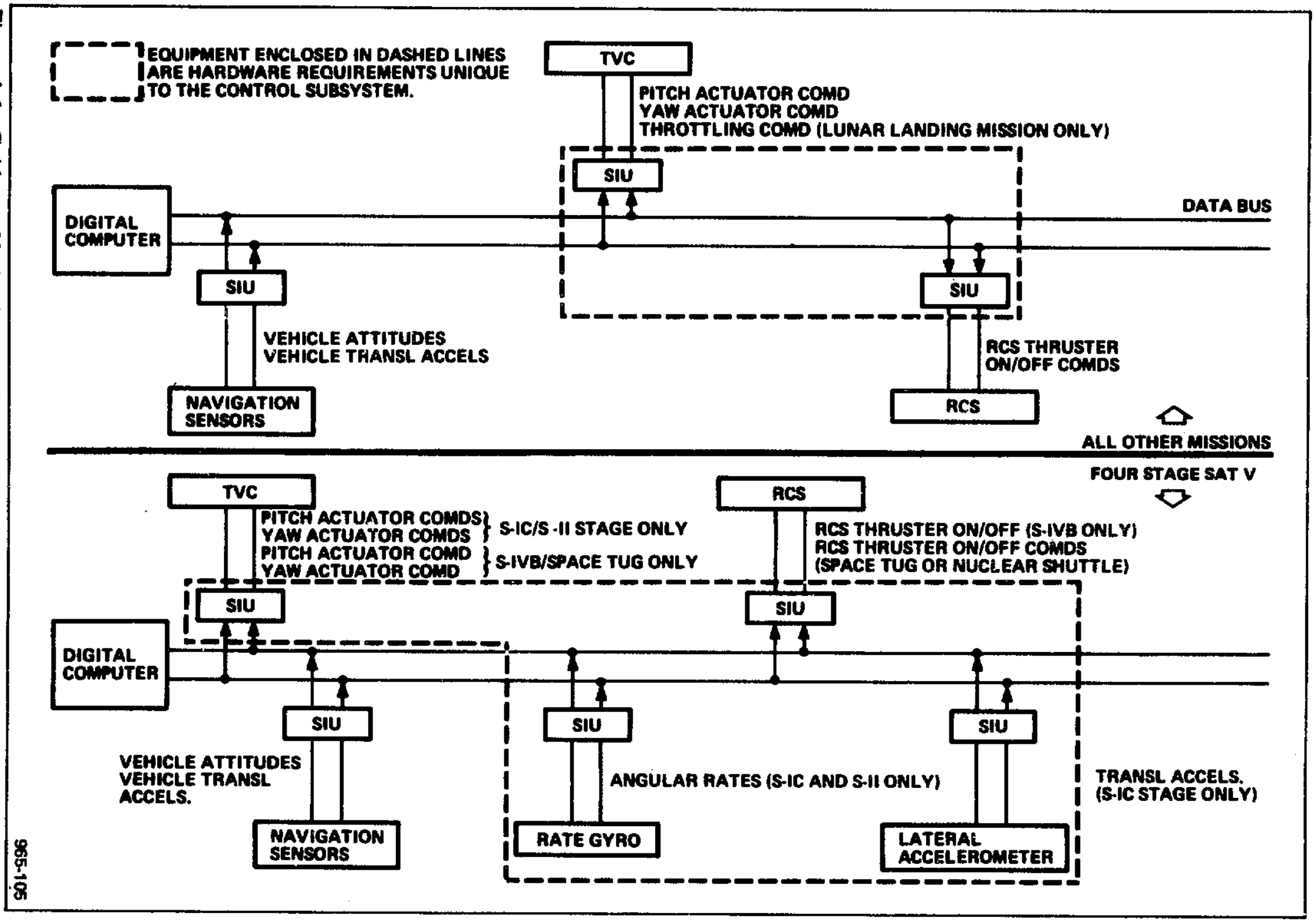


Figure 4-6. Guidance, Navigation and Control Layout for Tug Missions

Strenuous control requirements are met by employing the RCS for three-axis attitude control (Acceleration Nulling Mode). During this mode, it is desired that the gimbal engine thrust vector point through the vehicle center of gravity so that the disturbance angular accelerations due to gimbal engine thrust misalignment and vehicle center of gravity offsets are minimized.

The TVC law for the Acceleration Nulling Mode consists of two parts. First, the engine sign command is determined, and, second, the time required to null the disturbance acceleration is computed. The engine is then gimballed at a rate of 0.2 deg/sec in the acceleration nulling direction for the duration commanded. The drive direction is redetermined every 2 seconds.

The TVC subsystem controls the attitude about two axes and the RCS controls the attitude about the third axis when the Attitude Control Mode is employed. During this control mode operation, the attitude error and its first and second derivatives are driven to zero along the time optimal trajectory in the three-axis attitude phase space (Reference F-8, page 3.5-2). The output of the control law is actually a polarity, either plus or minus, or zero. Thus, in a plane the engine is commanded to gimbal in either a positive or negative direction at a fixed gimbal rate (0.2 deg/sec) or remain at the null position. The computation of the gimbal command is repeated every 0.2 sec. Between evaluations of the attitude control law, the engine moves at the fixed rate or remains at null position.

The primary advantages of the Attitude Control Mode are as follows:

- It allows reduction of RCS fuel consumption.
- It reduces the number of RCS engine firings for engine reliability purposes.

The LM engine control laws are not considered desirable for general space tug application. They were designed for an engine possessing very limited maneuvering rate capabilities to trim misalignment conditions and perform limited rendezvous, docking, and landing conditions.

(2) The CSM TVC law (Reference F-7) provides attitude control about the pitch and yaw axes. Roll control is provided by RCS. The attitude errors are transmitted to the digitally implemented compensation filters whose outputs are the commands to the engine gimbal servos. The control law is expressed as follows:

$$B_c = A_0 F(Z) \Delta\phi$$

where:

- B_c is the engine deflection command signal
- A_0 is the attitude control gain
- $F(Z)$ is the digital filter
- $\Delta\phi$ is the attitude error signal before filtering

Damping is inherent in the filter design for stability. The filter gains are adjusted to variations in thrust, vehicle center of gravity, and vehicle moment of inertia. This type of TVC control law is adequate for most space tug missions and mission phases.

(3) The S-IB stage TVC control law depends on attitude, attitude rate, and lateral acceleration feedback. The compensation filter implementations consist of electrical networks. This implementation is commonly known as analog filter implementation versus the digital filtering implementation used by the CSM. Lateral acceleration feedback is included to provide structural load relief due to vehicle bending during the S-IB stage boost through the max Q region in the earth's atmosphere. Filter gains are preprogrammed to insure stability and to provide satisfactory transient responses to control commands and disturbances. A similar control law will possibly be needed for the S-IC stage boost of the space tug four stage Saturn V missions. Lateral acceleration feedback may be required if the nominal control law does not provide adequate load relief.

The bending moment of a boost vehicle can be expressed as

$$M_B = K_\alpha \alpha + K_{B_E} B_E + \sum_{i=1}^{K} K_{\ddot{\eta}_i} \ddot{\eta}_i$$

where

- M_B denotes the total bending moment
- α denotes the angle of attack
- $\ddot{\eta}_i$ denotes the lateral acceleration of the i^{th} bending mode
- K_α, K_{B_E}, $K_{\ddot{\eta}_i}$ denote the appropriate coefficients

Lateral acceleration feedback tends to increase $\ddot{\eta}$ and to reduce α. For the critical station on Saturn V, the increase in $\ddot{\eta}$ would have been significant enough to nearly counter the reduction in α. Therefore, the reduction in the total bending moment that would have been achieved by including lateral acceleration feedback was considered too small to warrant including lateral accelerometer control sensors.

The TVC laws for the three stages of the Saturn V incorporate attitude error and attitude rate feedback only. The Saturn V control function is also implemented with analog filters. Lateral acceleration feedback is not presently required for Saturn V since the attitude/attitude-rate control law provides sufficient load relief.

(4) A modified disturbance acceleration nulling scheme may be used for the tug lunar landing mission. The purpose of the control law is to compute the change in the engine gimbal angle (ΔB_c) required to point the thrust vector through the space tug center of gravity. (The derivation of this angle is shown in Figure 4-7.) The engine deflection command would be incremented each computation cycle by ΔB_c. The engine command would be updated not more than perhaps every 2 seconds to allow engine deflection damping following the issuance of an engine deflection command change. The equation flow for the engine commands would be as follows:

$$B_{c_o} = \Delta B_{c_o}$$

$$B_{c_i} = B_{c_{i-1}} + \Delta B_{c_i} \quad i = 1,2,3,\ldots n$$

It should be noted that this scheme is similar to the Acceleration Nulling Mode employed by the LM. When the Acceleration Nulling Mode is selected, B_c is in effect divided by the engine gimbal rate to compute the time required for the engine gimbal to null the disturbance acceleration.

4.2.1.2.2 Space Tug TVC Law Requirements. The operational TVC attitude control law for the space tug will be dependent on the particular mission and mission phase requirements. Three versions of the TVC law, TVC1, TVC2, and TVC3, will be sufficient to encompass all missions and mission phases. These are as follows:

TVC1: $B_c = A_o F(Z)_\phi \Delta\phi$ (pitch or yaw)

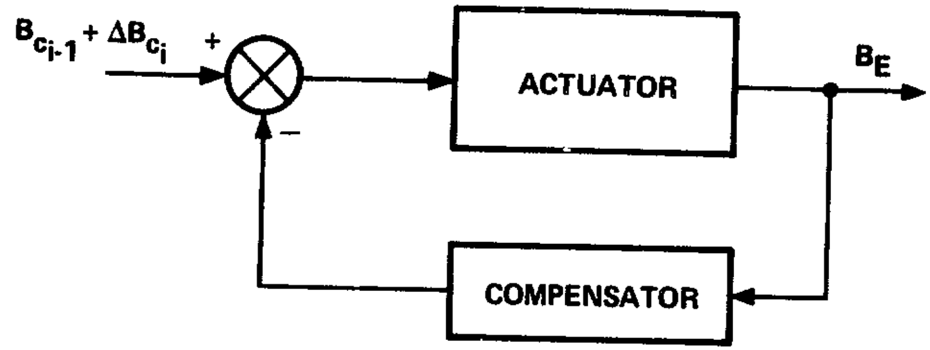
$B_c = A_o F(Z)_\phi \Delta\phi + A_1 F(Z)_{\dot\phi} \Delta\phi + g_2 F(Z)_{\ddot\gamma} \ddot\gamma$ (pitch or yaw)

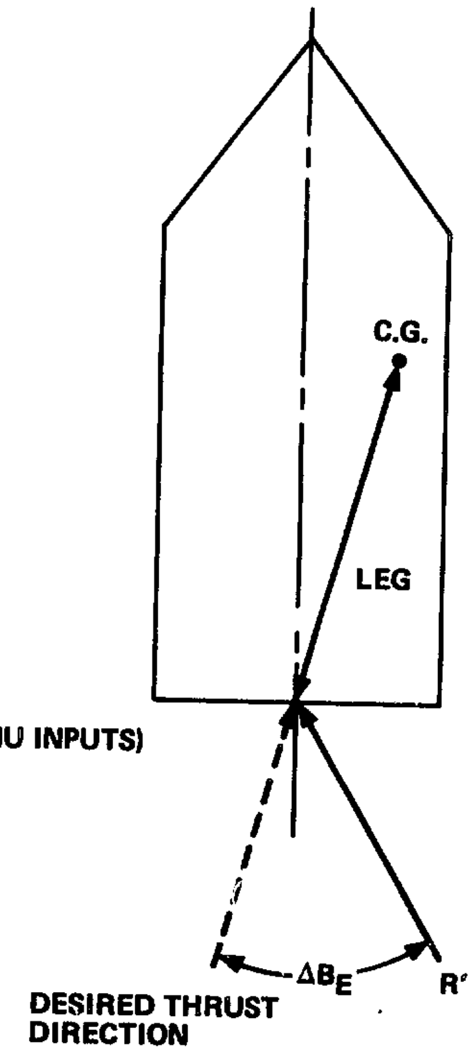
TVC2: $B_R = A_{oR} F(Z)_{\phi R} \Delta\phi_R + A_{1R} F(Z)_{\dot\phi} \Delta \dot\phi_R$ (roll)

TVC3: $B_{c_i} = B_{c_{i-1}} + \Delta B_c$ (pitch or yaw)

TVC1 (based on the CSM TVC law) will be the most commonly used of the three laws. It will be used for pitch and yaw attitude control during all thrusting missions and mission phases except during certain phases of the lunar landing mission and of the four stage Saturn V mission. The IMU will furnish the necessary attitude measurements needed to compute the engine deflection commands. Roll attitude control will be provided by RCS.

TVC2 (based on the S-IB stage TVC) is a strong candidate for the S-IC and S-II stage burn portions of the four stage Saturn V mission. However, g_2 will be zeroed throughout the S-II stage burn. Body-referenced lateral accelerometer and angular rate gyro inputs may be required to provide vehicle stability and load relief due to vehicle bending during S-IC stage burn. The accelerometer package and rate gyro package will likely be located in the aft section of the S-IVB stage instead of in the astrionic module. The sensors will thus be less susceptible to vehicle bending disturbances. TVC2 will be used to accomplish pitch, yaw, and roll attitude control. TVC1 will be used for pitch and yaw control during the S-IVB and space tug burn modes. Roll control will be provided by RCS.





- $\ddot{\phi}_d$ IS THE DISTURBANCE ACCELERATION PRESENT (FROM EQ (1))
- Ixx IS THE VEHICLE MOMENT OF INERTIA (PREDICTED,
- Leg IS THE DISTANCE FROM THE HINGE PIN OF THE GIMBAL ENGINE BELL (PREDICTED) TO THE SPACE TUG CENTER OF GRAVITY
- R' IS THE RESULTANT ENGINE THRUST (PREDICTED)
- ΔB_c IS THE ENGINE GIMBAL DEFLECTION REQUIRED TO NULL THE DISTURBANCE ACCELERATION (FROM EQ (3)
- $\ddot{\phi}_t$ IS THE TOTAL ANGULAR ACCELERATION PRESENT (ESTIMATED FROM IMU INPUTS)
- $\ddot{\phi}_{rcs}$ IS THE CONTROL ANGULAR ACCELERATION PRESENT (ESTIMATED)

EQ (1) $\ddot{\phi}_t = \ddot{\phi}_{r_{c_s}} + \ddot{\phi}_d$ (USED TO COMPUTE $\ddot{\phi}_d$)

EQ (2) $\ddot{\phi}_d Ixx = R' (\sin \Delta B_c)\, Leg$

EQ (3) $\Delta B_c = \sin^{-1} \ddot{\phi}_d \dfrac{Ixx}{R'\ Leg}$

965-106

Figure 4-7. A Modified TVC Disturbance Acceleration Nulling Scheme

TVC3 will provide disturbance angular acceleration nulling during descent and ascent phases of the lunar landing mission (refer to the previous section). Pitch, yaw, and roll control will be provided by the RCS.

A potential problem area associated with TVC1 deals with recovery from space tug transients occurring because of vehicle center of gravity offsets and thrust vector misalignments. This phenomenon generally occurs during engine thrust buildup following the engine start command. Thrust vector misalignment of the chemical engine should not be significant enough to pose a recovery problem. However, an excessive c.g. offset attributed to configuration or payload is not acceptable because, with limited engine deflection capability, sufficient control authority would not be available. After thrust has reached steady-state and attitude recovery has occurred, steering misalignment correction, a guidance function, will serve to keep the vehicle acceleration vector pointed in the desired direction. The steady-state engine gimbal angle will align the thrust vector through the vehicle center of gravity.

4.2.1.2.3 TVC Software Requirements. Of the tug missions, the four stage Saturn V mission (Figure 4-8) will impose the most strenuous TVC subsystem software requirements. Representative mission filter requirements* are shown in Table 4-5.

Refer to Table 4-6 for the estimated S-IC and S-II stage TVC computer requirements. The S-IVB stage and space tug TVC computer requirements will be assumed to be the same as the space tug TVC requirements listed in Table 4-1.

Refer to Section 3.1 for a summary of software requirements for the total control function for all space tug missions.

* For this study, the TVC subsystem solution rate will be assumed to be 10 samples per second for all stages.

Table 4-5.

STAGE	CHANNEL	FILTER ORDERS	NO. OF CONTROL GAIN STRAIGHT LINE SEGMENTS
S-IC	PITCH AND YAW ATTITUDE ERROR	6/6	4
S-IC	ROLL ATTITUDE ERROR	4/4	2
S-IC	PITCH AND YAW ANGULAR RATE	6/6	4
S-IC	ROLL ANGULAR RATE	4/4	2
S-IC	PITCH AND YAW LATERAL ACCEL.	6/6	3
S-II	PITCH AND YAW ATTITUDE ERROR	6/6	4
S-II	ROLL ATTITUDE ERROR	4/4	2
S-II	PITCH AND YAW ANGULAR RATE	6/6	4
S-II	ROLL ANGULAR RATE	4/4	2
S-IVB	PITCH AND YAW ATTITUDE ERROR	8/8	4
SPACE TUG	PITCH AND YAW ATTITUDE ERROR	8/8	2

Table 4-6. Four Stage Saturn V Mission, S-IC and S-II Stage TVC Computer Requirements

	NUMBER REQUIRED	MULT.	ADD/SUB	NON-ARITH	DATA	TOTAL MEMORY	EQUIV. ADDS	CALLS/SEC	EQUIV. ADDS/SEC
<u>S-IC STAGE</u>									
6/6 FILTERS (REFERENCE F-11)	6	13	13	48	29	618	678	10	6,780
4/4 FILTERS	2	9	9	32	20	140	154	10	1,540
Δ K	26	2	2	4	6	364	—	—	—
CHANNEL GAINS K	8		1	4	3	64	40	10	400
DECISIONS (REFERENCE F-12)	1			4	1	5	4	10	40
SIGNAL MIXING			6	12	10	28	18	10	180
LIMITERS	8		2	16	10	224	144	10	1,440
TOTAL (S-IC)		148	178	632	485	1,443	1,038	10	10,380
<u>S-II STAGE</u>									
6/6 FILTERS	4	13	13	48	29	412	452	10	4,520
4/4 FILTERS	2	9	9	32	20	140	154	10	1,540
Δ K	20	2	2	4	6	280	—	—	—
CHANNEL GAINS K	6		1	4	3	48	30	10	300
DECISIONS	1			4	1	5	4	10	40
SIGNAL MIXING			6	12	10	28	18	10	180
LIMITERS	6		2	16	10	168	108	10	1,080
TOTAL (S-II)		110	134	472	365	1,081	766	10	7,660

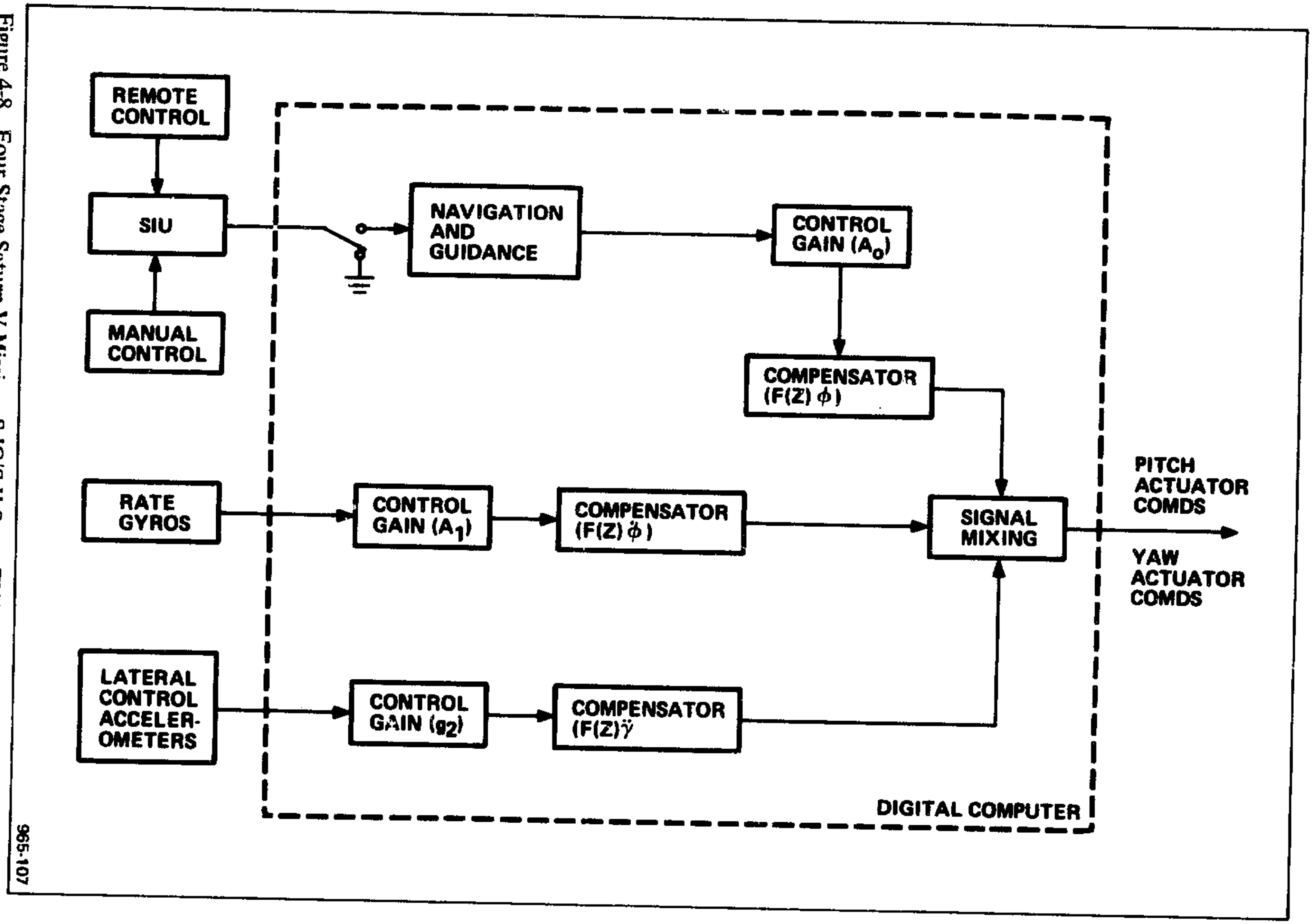


Figure 4-8. Four Stage Saturn V Mission, S-IC/S-II Stage TVC

4.2.2 RCS Thruster Sizing Considerations

The RCS engine thrust magnitude requirements for attitude control are dependent upon:

- Vehicle and payload characteristics
- Maximum and minimum angular acceleration requirements about each respective axis for specified mission maneuvers.
- Locations of the RCS thrusters.
- Number of thrusters firing for a commanded maneuver.

The RCS thrust magnitude requirements for translational control are dependent upon the following:

- Maximum and minimum acceleration requirements along each respective axis for midcourse maneuvers, docking, rendezvous, etc.
- Criticality of available fuel.
- Vehicle mass characteristics.
- Number of thrusters firing for a commanded maneuver.

The total number of RCS engines required to perform the desired attitude and translational control functions is dependent on factors which include:

- Redundancy requirements
- Thruster location – translational control requires no resultant moment about the c.g.; attitude control is effected by generation moments about the c.g. – therefore any sharing of engines between the two functions is dependent on engine location.

4.2.3 Manual and Remote Operation Requirements

4.2.3.1 RCS Attitude Control Requirements

For manned missions, capability for manual or remote override of the automatic control function will be provided. For manual override, the pilot will issue attitude rate commands to the control function (Figure 4-9). The attitude rate commands will be differenced from the actual body rates to form rate error inputs to the control law. Rate commands are especially desirable for pilot takeover since man is able to sense rate changes. For remote override, attitude or attitude rate commands will again be issued to the control function via the guidance function. Remote operation of the tug may require a television display system at the remote station and onboard the tug.

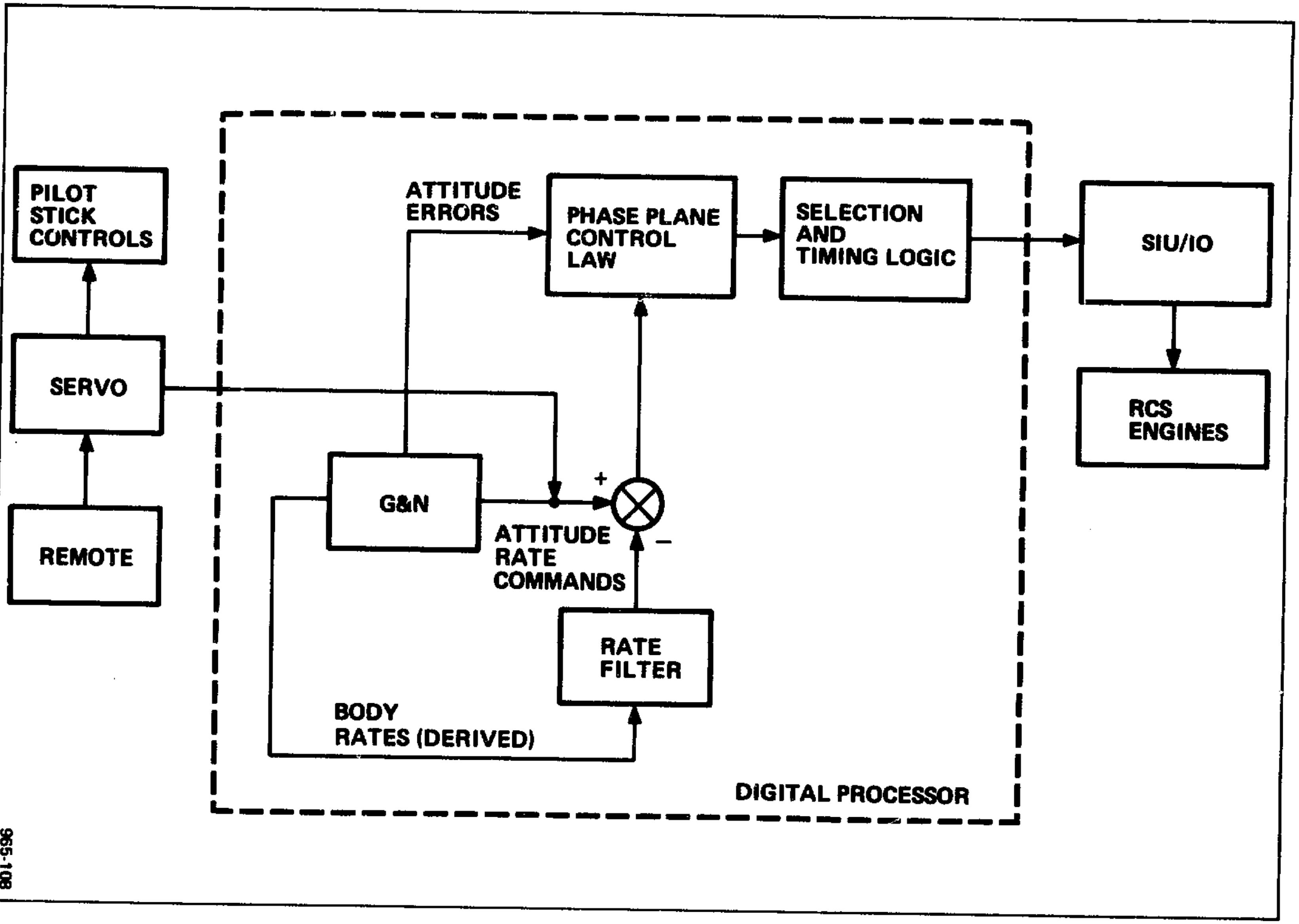


Figure 4-9. RCS Attitude Control

The pilot will have the option of providing either attitude rate or angular acceleration command inputs to the control computation when the manual control mode is selected. The attitude rate command will be similar to that discussed above. The angular acceleration command is more direct in that the pilot has a more direct link to the on-off logic of the RCS engines (refer to Figure 4-10).

4.2.3.2 RCS Translation Control Requirements

The pilot will have at least two options for selecting the type of translational control depending on the circumstance. The two primary options will be:

- The pilot will be able to select any particular set of reaction engines for translational control along any of three axes (acceleration control).
- The pilot will be able to select a desired ΔV along the axis of interest, and the engine selection and burn operation will be automatically performed.

4.2.3.3 Control Impact of Throttling Main Engine

The main engine will have throttling capability. The pilot will select a ΔV type control particularly for the lunar landing descent/ascent phase. The ΔV command will serve as a command input to the digital processor in which a routine will compute the desired thrust level. Acceleration control will also be available; that is, the pilot will have the option of a more direct thrust control.

The TVC attitude control subsystem will be affected by thrust level command changes. A thrust level reduction with no alteration in the control gains will result in less control authority which may or may not result in an uncontrollable condition. Control gains which are adaptive to thrust level commands will provide the necessary compensation.

A throttleable engine requirement has been identified only for the lunar landing mission (Reference Appendix E, Guidance Analysis and Implementation).

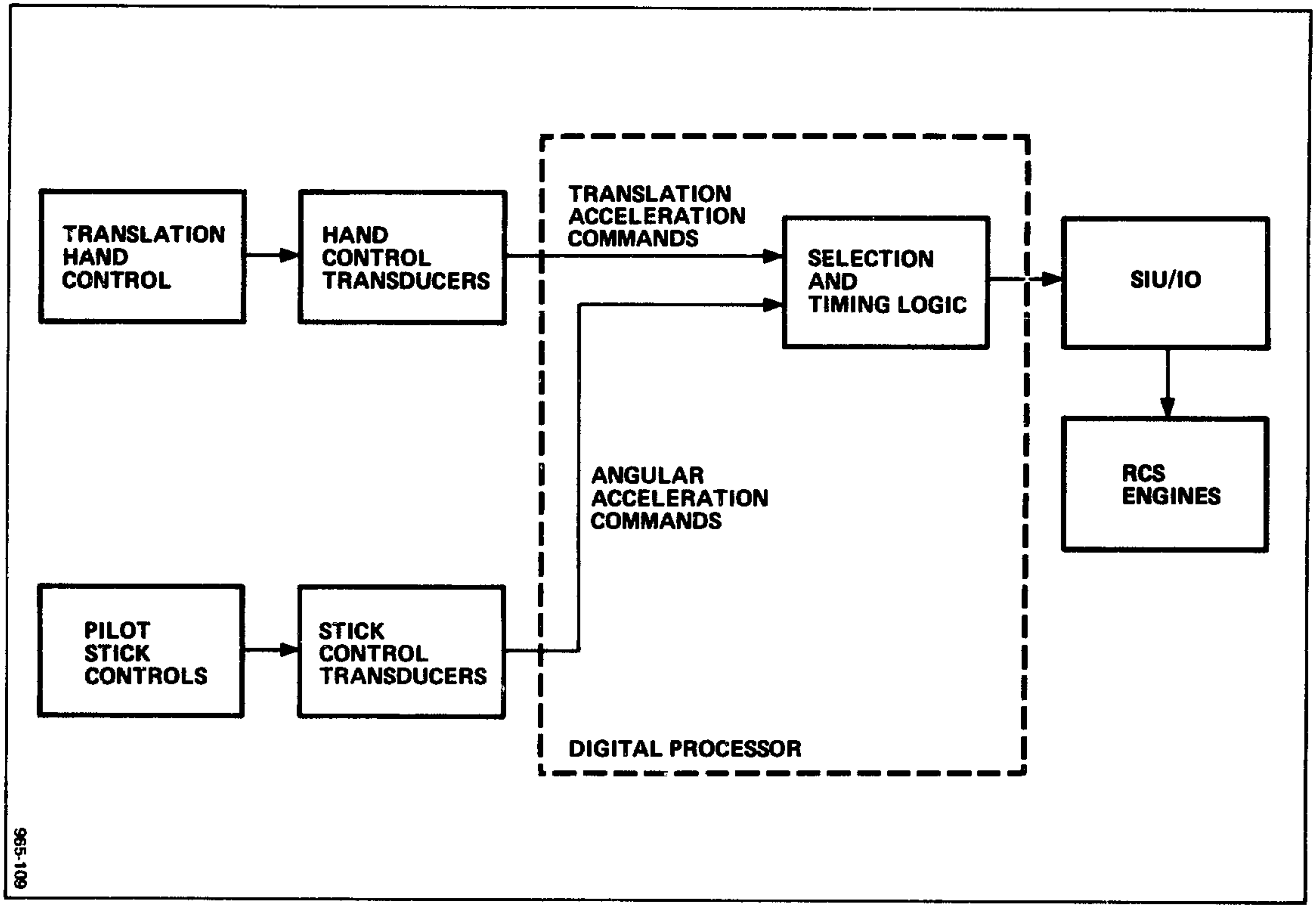


Figure 4-10. Manual RCS Attitude and Translational Acceleration Control

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APPENDIX G

SPACE TUG ELECTRICAL POWER
ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

This appendix presents the results of preliminary studies for space tug electrical power requirements and layouts. Primary emphasis is given to the synchronous orbit mission as a baseline, but the entire family of candidate space tug missions is treated on the basis of commonality of requirements with the baseline mission. Additional data is presented on a power system option for the lunar landing mission which yields a significant cost reduction but also entails a substantial weight penalty.

A major portion of the study effort was devoted to identification of electrical load requirements and ultimately the derivation of a power profile for the synchronous orbit mission. This was accomplished through the preparation of work sheets which included mission by mission equipment lists, power per component and identification of active, standby and inactive quantities of each component. Data obtained in this manner was then compiled and time correlated with the mission timeline to generate the power profile for the first tug for the synchronous orbit mission. A time weighted average power was then derived from the profile and, due to relatively minor variations between mission astrionic equipment lists, was used as an estimating base for coarse sizing of the power system for remaining missions.

The spectrum of space tug missions addressed in this study are divided into three classes, (1) unmanned expendable (2) unmanned reusable and (3) manned reusable. In the interest of achieving acceptable reliability while minimizing cost for the expendable and unmanned reusable tugs the modular concept illustrated in Figure 1-1 was adopted. Although detailed reliability analysis has not been performed on this power system approach it is expected to be capable of meeting the fail safe and repair criteria for expendable tugs and the fail operational/fail safe criteria on manned reusable vehicles.

A "quick look" assessment was made of the additional electrical loads imposed by the crew module and propulsion module equipment in order to determine the impact of an integrated power system located in the tug astrionic module.

The power distribution concept developed for the synchronous orbit tug follows the modularity concept of Figure 1-1. The emergency power supply which would be used only on manned missions has been included to show the routing of power to those critical astrionic loads which would be needed to return the crew safely to an orbital base.

2.0 STUDY GUIDELINES AND GROUNDRULES

The following guidelines and groundrules apply to the space tug electrical power study effort:

1. Storage phases of the tug missions will not be included in the power and net energy calculations for any missions other than the lunar landing mission pending further definition of storage load magnitudes and duty cycles. The storage phase will be treated as a separate mission and will be analyzed in subsequent study tasks.

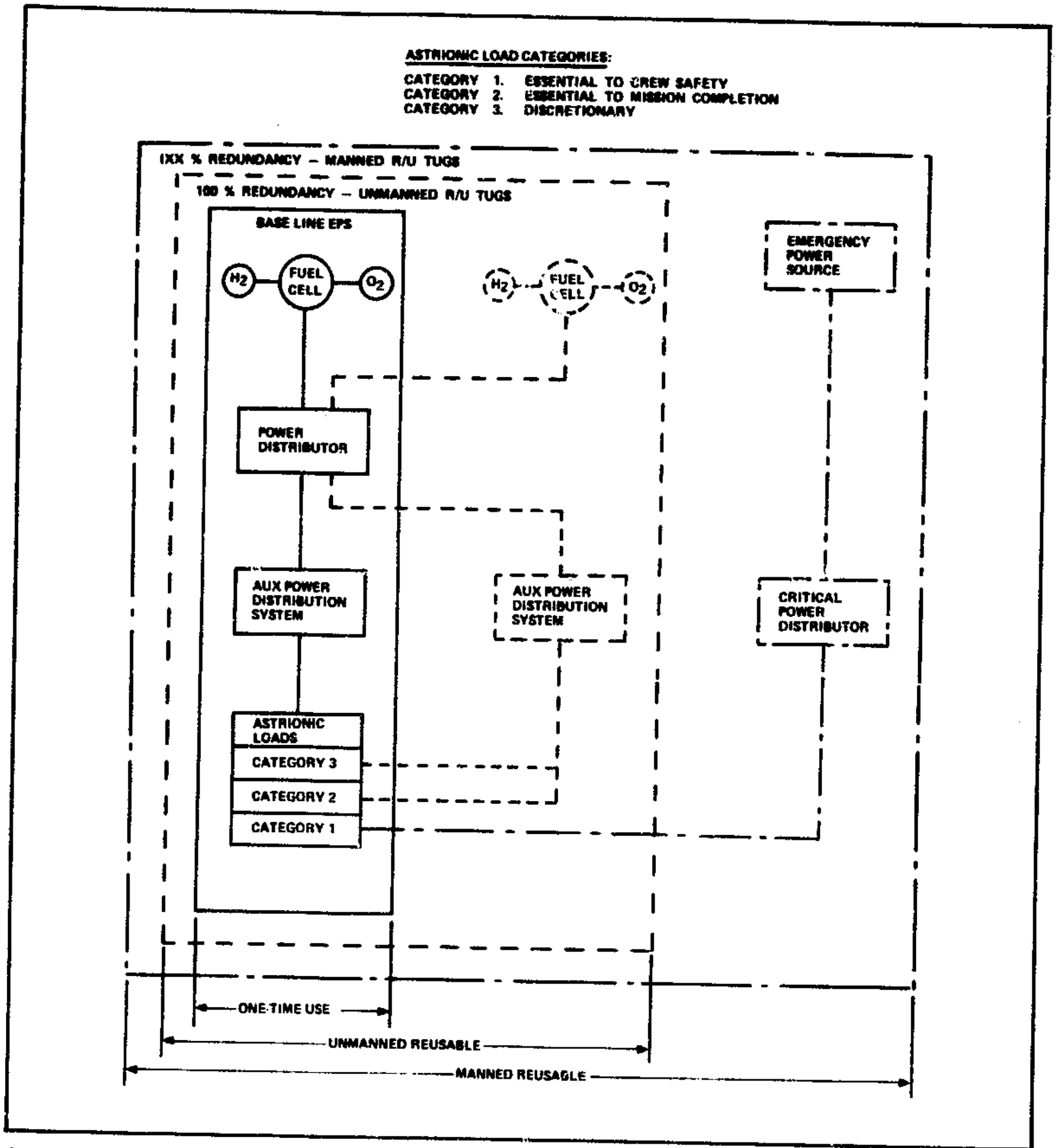


Figure 1-1. Modular Power Concept

2. The power system will be sized for the total astrionic system and a "quick look" made at an integrated power system for the total tug.

3. Power and environmental conditioning may be provided to the space tug astrionics by a space element when the tug is docked with the element.

4. As a minimum, equipment will be designed for a fail safe and repair operation with a fail operational/fail safe condition on equipment required for crew safety.

5. Crew module backup systems required after separation from the astrionic module are not considered in this study.

6. Between missions, maintenance will be performed as required to upgrade the reliability to the required level. During the mission, maintenance shall be limited to switching in redundant systems.

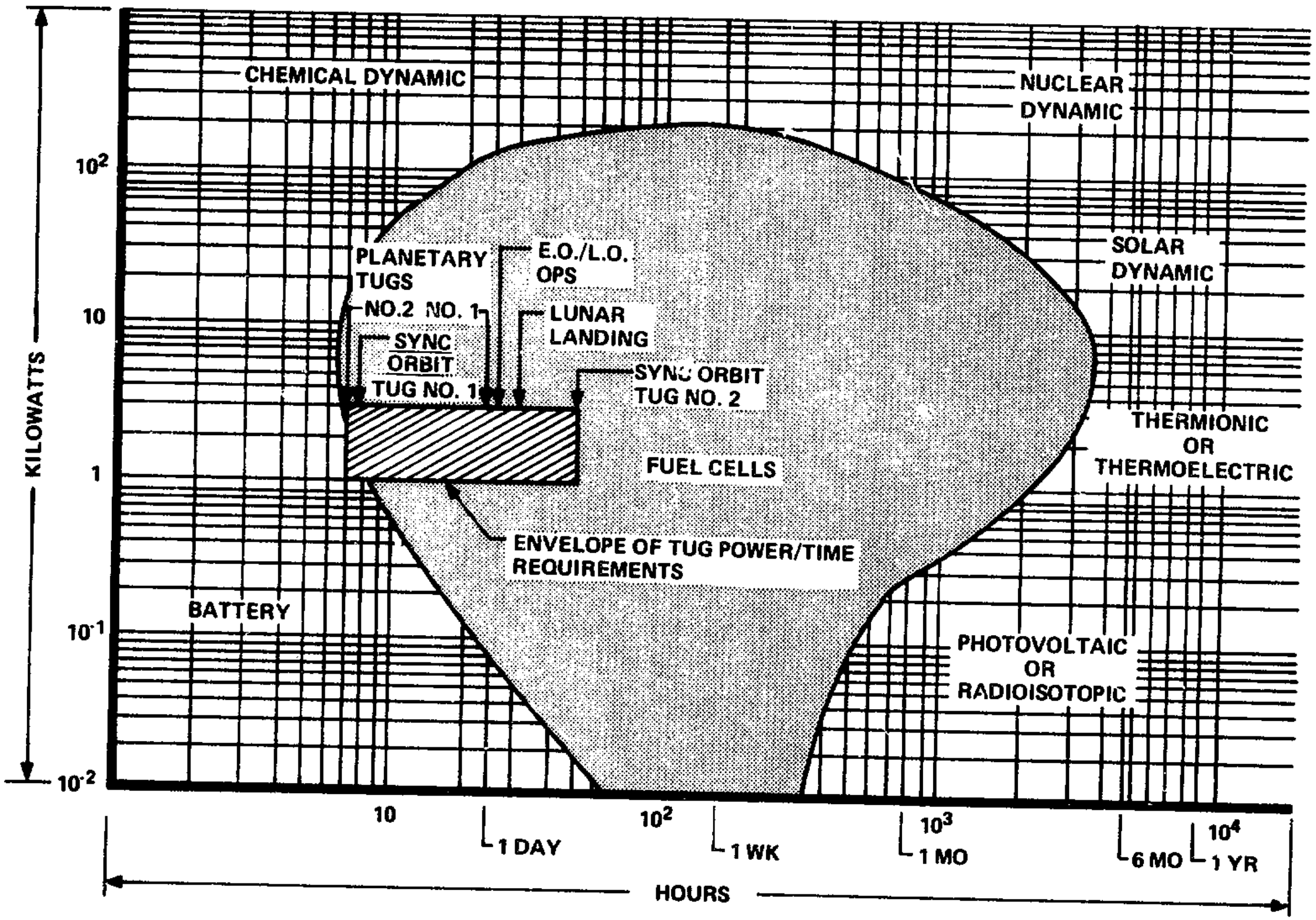
3.0 SUMMARY OF RESULTS

The conclusion of this phase of the electrical power system study is that common fuel cell power systems varying only in the number of fuel cells, size of reactant tanks and addition of an emergency battery power supply can effectively support the entire spectrum of candidate missions for the space tug, as defined for this study. The choice of fuel cells for this family of missions is supported by identification of space tug mission power/time requirements on the power technology map of Figure 3-1 (Reference G-1). This map is based on the assumption that the power system is provided with the entire mission reactant requirement at liftoff. The region of optimum weight application for fuel cells will be greater still if the concept of on-orbit resupply (e.g., prior to entering storage phases) is applied to the tug missions.

The choice of fuel cells for the synchronous orbit baseline mission is based on consideration of the factors presented in the selection matrix, Figure 3-2. Most of these factors apply equally well to the other space tug missions because of equivalent power levels and a relatively narrow range of active phase time durations. The factors shown were weighted from 0 to 10 depending on the relative importance of each. For example, low weight and availability of hardware were considered very important and thus were weighted at a 10 level. The generation of a useful by-product was considered less important and thus weighted at a 4 level. These weighting factors are shown at the top of Figure 3-2. Each of the electrical power source/conversion candidates were compared for each of the weighted factors and assigned a value relative to the weighting factor. For instance, for the low weight factor, the fuel cells are relatively light, therefore, it is assigned a high value, 7. The batteries are heavy and are assigned a low value, 1. Each of the candidates are compared in this manner. The totals show that the fuel cells receive the highest total, and they were selected as the optimum candidate for electrical power generation, with batteries or fuel cell/battery combinations as other advantageous candidates.

For the lunar landing mission, power requirements are unique in that they are characterized by (1) high energy, short time duration power demands imposed by the orbit-to-surface and surface-to-orbit flight phases followed by extended periods of low power storage operation and (2) restraints imposed by operation in the lunar surface environment such as the extended periods of darkness which tend to preclude consideration of power systems which are dependent on solar energy. For this reason, an additional power option (fuel cells and rechangeable batteries) is presented as a cost reduction option.

The recommendation of fuel cells as the primary power source for the space tug is in keeping with the trend now in evidence for contemporary space transportation systems. Early phase study results on the space shuttle orbiter, which is characterized by mission times in the order of seven days and power levels comparable to those for the tug (with crew module support), have advocated fuel cells as the best primary source technology with availability in the 1972-73 time frame. A common technology for the space tug and other transportation systems would yield significant benefits in distributing of power system development costs.



(REF. G-1)

Figure 3-1. Region of Desirable Fuel Cell Utilization (Weight Optimized)

Figure 3-2. Power Technology Selection Matrix

ELECTRICAL POWER SOURCE/CONVERSION CANDIDATES		AVAILABILITY (10)*	LOW WEIGHT (10)*	LOW INITIAL COST (9)*	REUSE POTENTIAL (8)*	NO ORIENTATION CONFLICT (8)*	RELIABLE FLIGHT PROVEN (7)*	MINIMUM SAFETY HAZARD (6)*	MAXIMUM EASE OF INTEGRATION (5)*	MAXIMUM CONVERSION EFFICIENCY (5)*	REFURBISHMENT (5)*	USEFUL BY PRODUCT (4)*	TOTAL
ENERGY SOURCE	CONVERSION TECHNOLOGY												
CHEMICAL	PRIMARY BATTERY	10	1	9	0	8	7	5	5	4	5	0	54
	FUEL CELL	10	7	6	7	8	5	4	3	5	2	4	61
	FUEL CELL/SEC BATT	10	2	8	6	8	4	4	3	3	1	4	53
SUN	SOLAR CELL/SEC BATT	10	8	4	6	1	6	3	2	2	3	0	45
NUCLEAR (ISOTOPE OR REACTOR)	THERMO ELECTRIC	7	7**	1	8	3	3	2	4	1	4	0	40
	THERMIONIC	2	9**	4	8	5	1	2	4	2	4	0	41
	BRAYTON CYCLE	5	7**	3	6	6	2	1	3	4	2	0	39
	RANKINE CYCLE	4	8**	2	6	6	1	1	3	3	2	0	36

*WEIGHTED FACTORS
**DOES NOT INCLUDE SHIELDING

4.0 DETAILED ANALYSIS

4.1 SYNCHRONOUS ORBIT MISSION

The severe payload weight restraint for the synchronous orbit tug mission dictates weight minimization as the dominant trade factor for the power system. Secondary selection factors include the need for uninhibited maneuverability (docking, satellite retrieval, etc.) and freedom from aerodynamic drag penalties when in low earth orbital storage.

4.1.1 Power Requirements and Source Selection Criteria

Power requirement analysis for the synchronous orbit mission was carried out through the use of specially prepared work sheets which identified (1) astrionic equipment complement by mission and (2) electrical load per component. From these lists equipments were categorized as steady state or intermittent loads through consultation with the responsible subsystem engineers. Intermittent or cyclic loads were, then, further analyzed in terms of function and usage by mission phase to generate the power profile for the synchronous orbit reusable second tug, shown in Figure 4-1.

From the power profile a time weighted average power of 837 watts was calculated as follows:

$$\text{Pavg} = \frac{t_1}{T} \times P_1 + \frac{t_2}{T} \times P_2 + \cdots\cdots \frac{t_n}{T} \times P_n$$

where:

T	=	total active mission time (hours)
t_1	=	time at power level P_1 (hours)
t_n	=	time at power level P_n (hours)
Pavg	=	time weighted average power (watts)

Based on 60 hours total mission time for the synchronous orbit second tug:

$$\begin{aligned}\text{Pavg} = \; & 2/60 \times 1040\text{w} + 1/60 \times 940\text{w} + .5/60 \times 840\text{w} \\ & + .14/60 \times 900\text{w} + 6/60 \times 760\text{w} + .07/60 \times 900\text{w} \\ & + 4/60 \times 760\text{w} + 6/60 \times 960\text{w} + 24/60 \times 800\text{w} \\ & + .14/60 \times 900\text{w} + 6/60 \times 760\text{w} + .07/60 \times 900\text{w} \\ & + 8/60 \times 980\text{w} + 2/60 \times 740\text{w}\end{aligned}$$

$$\text{Pavg} = 836.96 \approx 837 \text{ watts}$$

Selection of a power source to satisfy the mission power requirements identified in Figure 4-1 is based on the following criteria:

- Weight – compatible with synchronous orbit payload capability
- Mission Compatibility – non-interference with primary mission objectives (i.e., docking, satellite rendezvous, EVA)

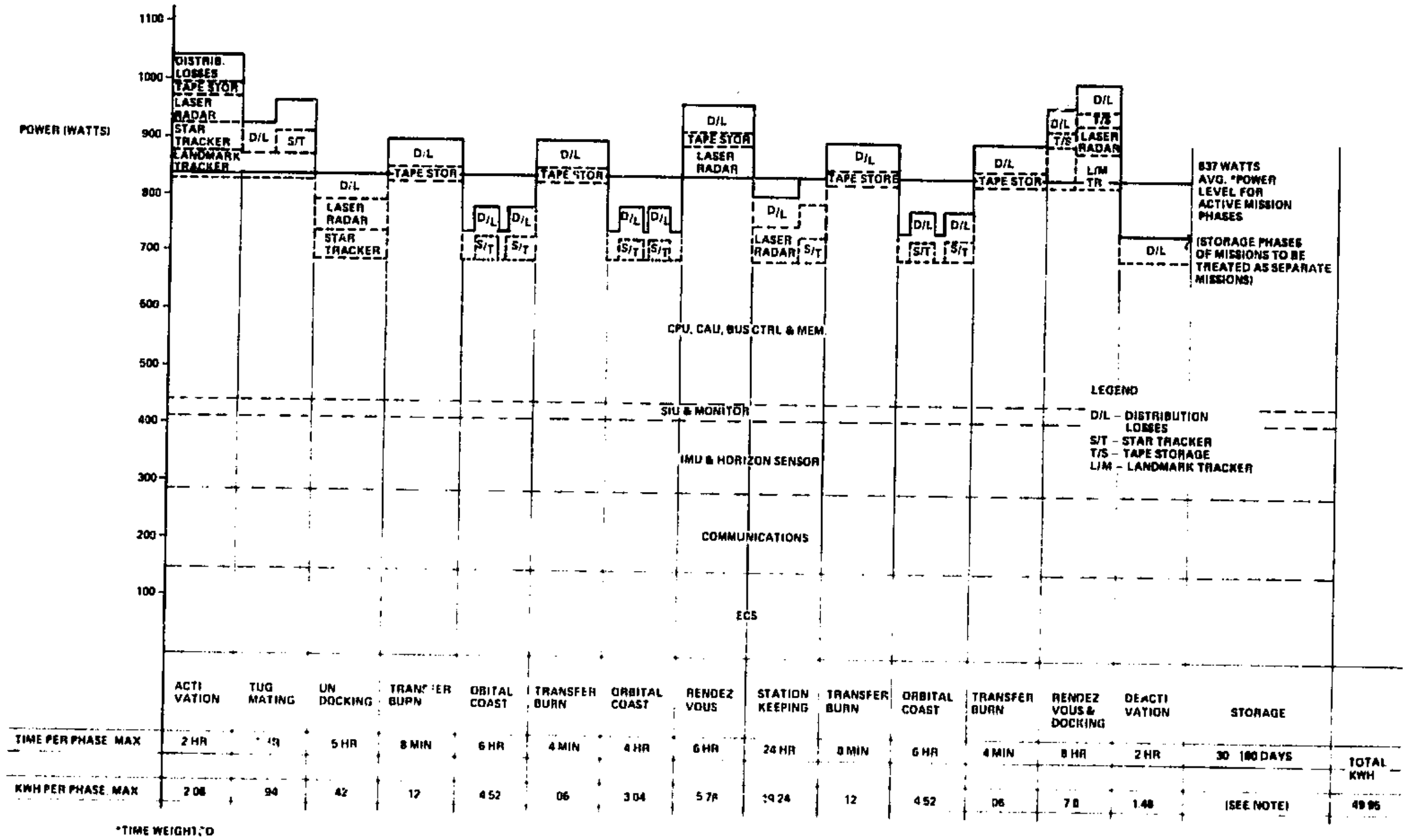


Figure 4-1. Power Profile – Sync Orbit Reusable 2nd Tug

- Availability – technology available in 1972-73 time period
- Autonomy – independent of other space tug modules and other space vehicles except for resupply and refurbishment from another space element
- Cost – cost competitive with other applicable power source technologies but as a secondary consideration to low weight

4.1.2 Applicability of Power Source Technologies

4.1.2.1 Solar Array/Battery System

A solar array/battery system is the lightest available power source for the synchronous orbit mission, as seen from the power source weight comparison, Figure 4-2 (Reference G-2). This system, however, is considered to impose unacceptable penalties in terms of mission compatibility and was rejected primarily on the following factors:

- Special vehicle orientation is required to obtain the required power with reasonable array area.
- External appendages (array panels) would interfere with docking maneuvers, EVA and communications antenna patterns.
- Solar array aerodynamic drag during low orbit storage is significant. It is estimated that 1700 pounds of propellant is required to maintain the 100 nm orbit for the solar array area required to generate one kilowatt of power (Reference G-3).

4.1.2.2 Primary Battery System

A primary battery system is not considered applicable for the synchronous orbit mission due to the following principal factors:

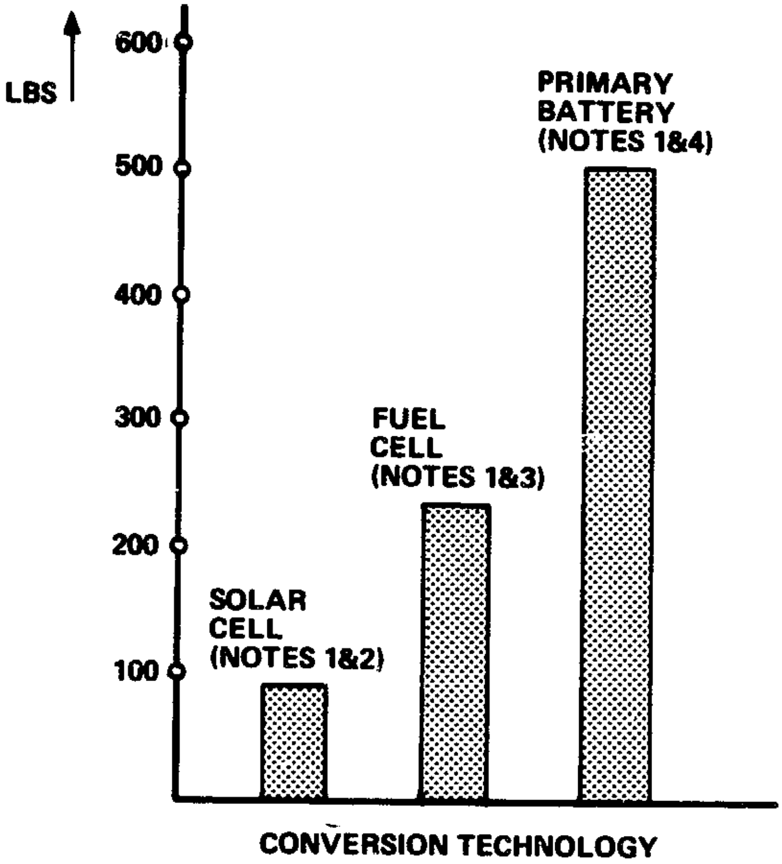
- Excessive weight
- Need for battery replacement after each round trip to synchronous orbit

On the basis of a 60 hour mission at the 840 watt average power level a redundant primary battery system using silver-zinc batteries at an energy density of 100 watt-hrs per pound would weigh 1008 lbs. calculated as follows:

$$W_B\text{, Battery Weight} = \frac{\text{Total Mission Energy}}{\text{Energy Density}}$$

$$W_B = \frac{60 \text{ hrs.} \times 840 \text{ watts}}{100 \text{ watt-hrs/lb}} = \frac{50{,}400 \text{ w-h}}{100 \text{ w-h/lb}}$$

$$W_B = 504 \text{ lbs} \times 2 \text{ (redundancy)} = 1008 \text{ lbs}$$



NOTES:

1. POWER SOURCE ONLY – WEIGHT OF POWER CONDITIONING & DISTRIBUTION EQUIPMENT NOT INCLUDED.

2. SILICON PANEL MOUNTED CELLS ORIENTED TO ± 15°: NiCd BATTERIES.

3. INCLUDES WEIGHT OF REACTANT & TANKS FOR SYNC. ORBIT MISSION.

4. Ag-Zn BATTERIES @ 100 WATT-HRS PER POUND.

(REF. G-2)

Figure 4-2. Non-Redundant Power Source Weight Comparison – Sync Orbit Mission (60 Hr. Duration)

This weight of batteries alone added to the weight of other power system components (cabling, distributors, etc.) would amount to an estimated 1300 pounds for the power system which is considered excessive for the synchronous orbit mission.

4.1.2.3 Fuel Cell System

As shown in Figure 3-1, the envelope of power-time products for the space tug missions falls almost entirely within the region of desirable fuel cell utilization. In addition, fuel cells represent the next lightest power source to solar arrays as indicated by the weight comparison of Figure 4-2. Fuel cells thus appear to best satisfy the selection criteria established for the synchronous orbit mission by virtue of the following factors:

- Weight – lighter than any source except solar cells
- Mission Compatibility – non-position sensitive and free of external appendages which could interfere with mission operations
- Reliability – suitable for manned missions of longer duration as demonstrated on Apollo and Gemini flights
- Availability – available with little or no development risk due to past program experience
- Autonomy – free of any dependence on external elements except for periodic maintenance and resupply of expendables
- Cost – lowest cost system with acceptable weight, as indicated in Figure 4-3 (Reference G-2).

4.1.2.4 Nuclear System

Nuclear systems are not appropriate for application to relatively low power, short time duration missions such as those considered in this study. They were eliminated from further consideration for the space tug based on the following factors:

Availability – very questionable due to development lag resulting from lack of assigned missions

High Cost – isotope costs vary from as low as \$24 (Reference G-4) per thermal watt for Strontium 90 (very hazardous) to \$500 (Reference G-4) per thermal watt for Plutonium 238 (relatively safe) but due to low efficiency of conversion devices the ratio of thermal power to electrical output power is approximately 20 to 1. Thus, the isotope cost alone for a Pu 238 fueled system using thermoelectric conversion and rated for a 1 KW electrical output would be on the order of ten million dollars.

Safety – the use of either radioisotope or nuclear reactor heated systems can result in severe radiation hazards and require extremely heavy shielding of the power source, particularly for manned missions.

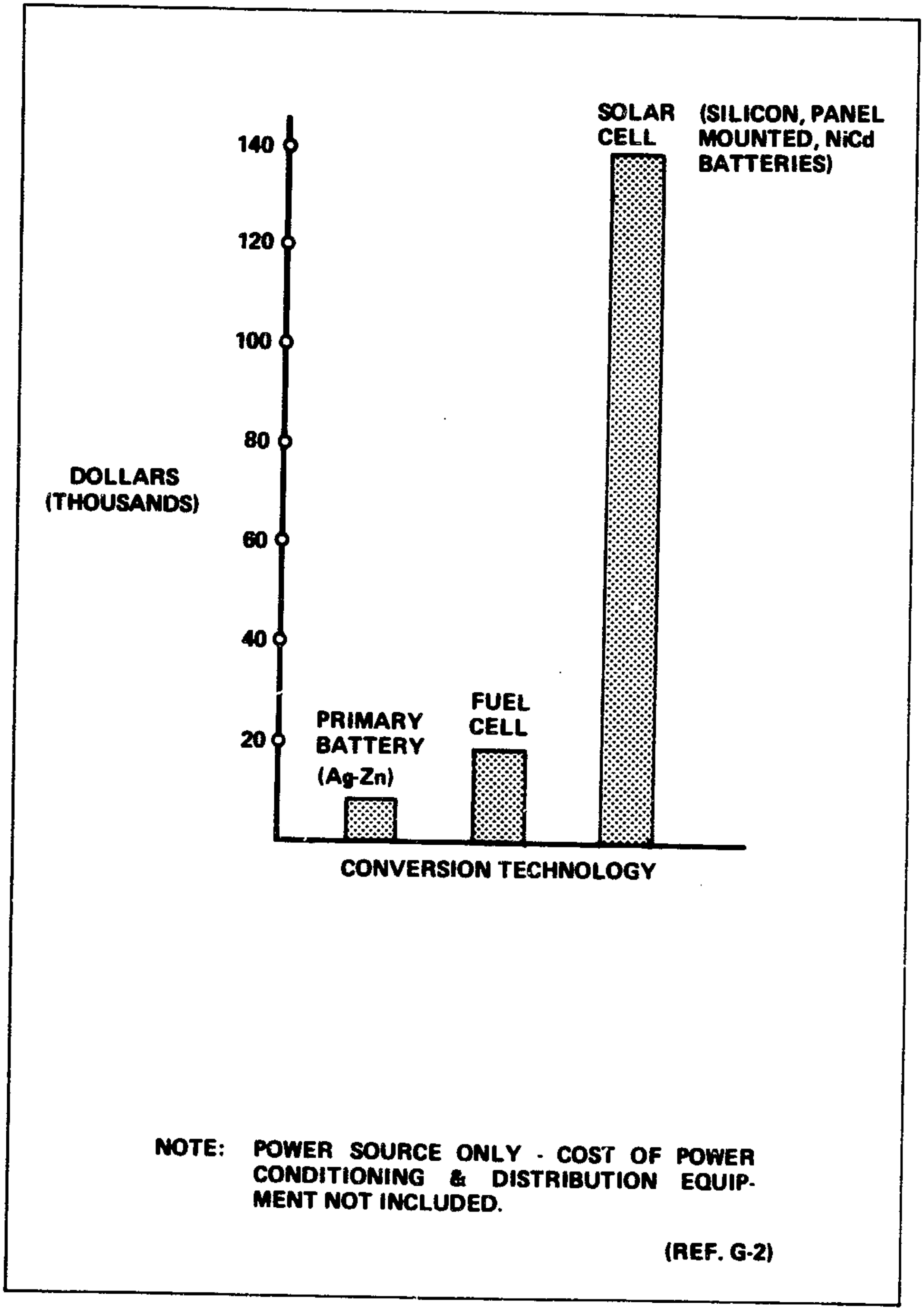


Figure 4-3. Non-Redundant Power Source Cost Comparison – Sync Orbit Mission (60 Hr.)

4.1.3 Conclusions

From this review of available power source technologies it is concluded that a fuel cell system is the optimum choice for the synchronous orbit mission. Other power sources reviewed, such as solar arrays, primary batteries, and nuclear systems, have serious disadvantages due to (1) excessive weight (2) lack of compatibility with mission objectives or (3) lack of availability in the specified 1972-1973 time frame.

4.1.4 Preliminary Design – Fuel Cell System

Based on the above conclusion that a fuel cell system best fulfills the requirements of the synchronous orbit space tug mission, preliminary design data for such a system was developed for the astrionic module power source.

4.1.4.1 Fuel Cell Sizing

In consideration of the 840 watt average power level derived from the power profile, Figure 4-1, a fuel cell of nominal one kilowatt capacity was considered initially. Several factors such as the preliminary nature of astrionic load data and the potential need for crew module and propulsion module support led to the conclusion that more growth margin in the basic Electrical Power System (EPS) building block should be allowed for initial sizing. Thus, the Allis-Chalmers 2 kilowatt, gas/liquid cooled fuel cell module, Figure 4-4, was chosen as the focus for preliminary design. This fuel cell has undergone intensive development since 1961 under various contracts from NASA (MSFC and MSC) specifically for aerospace application and has received considerable attention in studies performed for the USAF. This fuel cell offers a very favorable power-to-weight ratio (85 lb/KW) and low reactant consumption.

Following are salient characteristics of the Allis-Chalmers 2 KW, capillary matrix fuel cell (see Reference G-5):

Power Output/Module (nominal watts)	2000
Voltage (volts dc)	29
Weight (lbs/module)	169
Size (overall-inches)	16 x 21 x 32
Reactants	H_2O_2
Parasitic Power (watts)	100
Specific Energy (approx. wt-hrs/lb)	900*
Thermal Control Medium	Stack-gas cooled (vehicle coolant used to cool gas)

* Weight of reactant tankage and tankage components not included.

4.1.4.2 Reactant Requirements

Using the active mission average power level of 840 watts and mission time of 60 hours the total number of kilowatt-hours to be supplied by the fuel cell is:

KWH = 60 x 0.840 = 50.5 kilowatt-hours

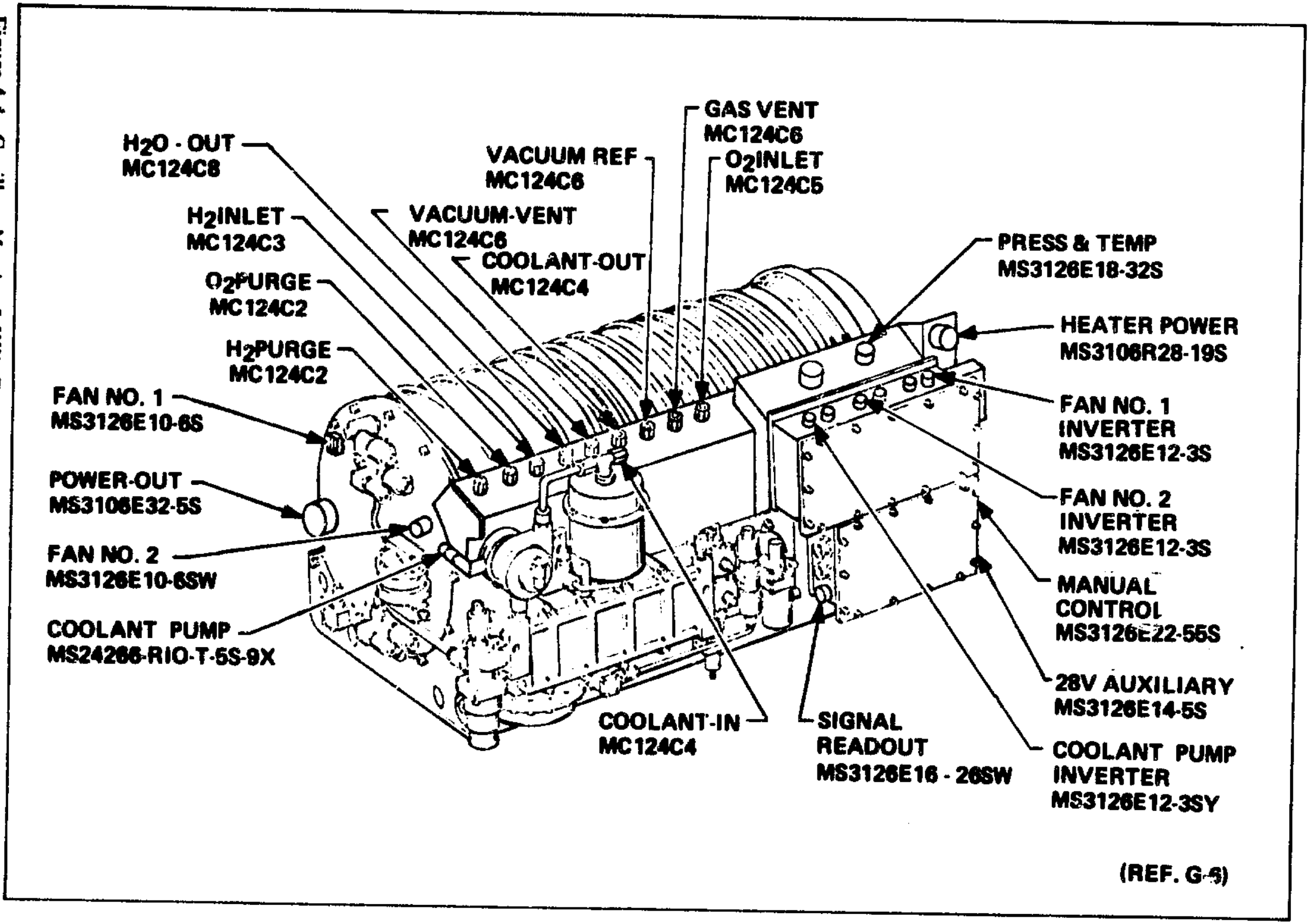


Figure 4-4. Capillary Matrix (2 KW) Fuel Cell Fluid and Electrical Interface Connections

Applying the specific reactant consumption rate of 0.83 lbs/KWH (at 1.5 gross power output and 100 days operation time) yields:

Weight of Reactant W_R = 0.83 lbs/KWH x 50.5 KWH = 41.8 lbs

Allowing 10% for reactant boiloff and 10% contingency gives a total reactant weight for the active mission of 50.4 lbs. Since the reactant is consumed in the ratio of 8 parts oxygen to 1 part hydrogen the weight of the individual reactants is:

W_{O2} = (8/9) 50.4 = 44.8 lbs of O_2
W_{H2} = (1/9) 50.4 = 5.6 lbs of H_2

4.1.4.3 Reactant Tank Sizing

Assuming subcritical cryogenic storage of the reactants the required tank weight is calculated from:

(1) lbs of H_2 x 1.19 = lbs of H_2 tank (Reference G-7)
(2) lbs of O_2 x 0.30 = lbs of O_2 tank (Reference G-7)

Substituting in (1) and (2):

5.6 x 1.19 = 6.65 lbs of H_2 tank
44.8 x 0.30 = 13.45 lbs of O_2 tank

Tank volume is calculated from:

(3) lbs of H_2 x 0.25 = cu. ft of H_2 tank (Reference G-7)
(4) lbs of O_2 x 0.018 = cu. ft of O_2 tank (Reference G-7)

Substituting in (3) and (4):

5.6 x 0.25 = 1.4 cu. ft of H_2 tank
44.8 x 0.018 = 0.81 cu. ft of O_2 tank

4.1.4.4 Emergency Battery Sizing (Manned Missions)

Sizing of an emergency battery, which would be applicable to manned missions only, is based on the following tabulation of critical loads required to return the crew safely to a space station or other space element:

Astrionic Load	Power (watts)
CPU (computer)	80
Bus Control Unit	50
Memory	100
Unified S-Band Equipment	107
SIU/IO	20
Mass Storage	15

Astrionic Load (Cont)		Power (watts)
Laser Radar		30
IMU (Guidance System)		200
USB Antenna Control		10
Command Decoder		10
	Total	622

Using 19 hours, the maximum flight time required to reach a point of safety (occurs in the lunar landing mission), the energy required from the emergency battery is:

E = 19 Hrs x 22 Amperes (622 watts @ 28 volts)
E = 418 Ampere hours

The silver-zinc LEM descent battery illustrated in Figure 4-5 (see Reference G-8) is typical of the size and type battery required to furnish this emergency power. The battery could be subjected to deep discharge (85%) without detectable voltage degradation because of the relatively few discharge cycles it would see in such emergency service. The 30 day wet stand life of the battery could be extended by design modification if this stand life proves incompatible with tug maintenance and repair procedures.

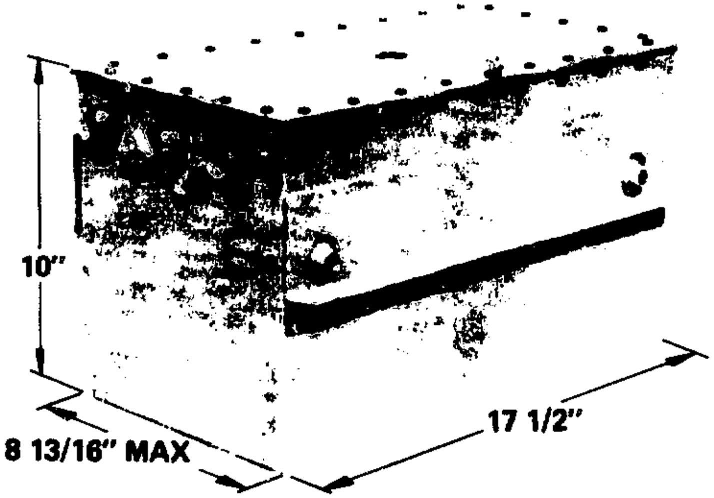
4.1.4.5 Distribution and Control Concept

The distribution scheme for the synchronous orbit second space tug is shown in Figure 4-6. In the redundant system shown, both 2 KW fuel cells feed a common power distributor which, in turn, supplies high and low current auxiliary distributors in the "A" and "B" load distribution subsystems. A bus tie switch allows a single fuel cell to feed the entire astrionic subsystem in the event of failure of one fuel cell. In normal operation the system would be isolated (bus tie open), and the fuel cells would operate in a load sharing mode to reduce electrical stress and extend fuel cell operating lifetime. As noted previously, the components shown in dashed lines are applicable to manned missions, only, and provide for emergency power to critical loads in the event of simultaneous failure of both the "A" and "B" fuel cell systems (e.g., meteriod puncture of both tank sets). The switching scheme provides for complete isolation of the critical load bus from the normal bus distribution system to protect the emergency supply from faults occurring in buses or wiring of the normal system.

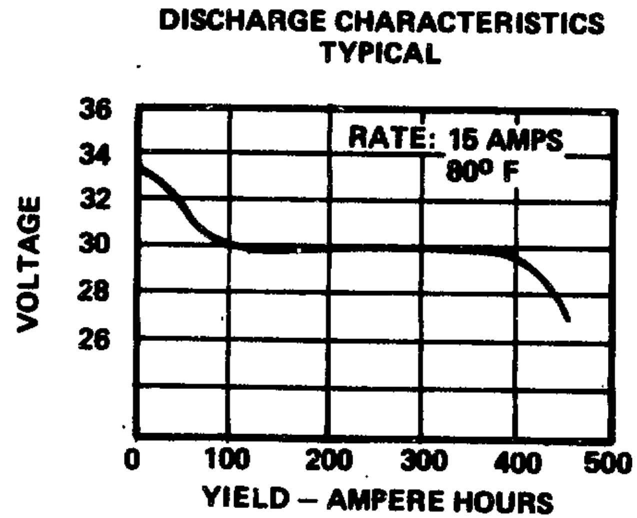
4.1.4.6 Power Weight Summary

The power supply and power distribution weight summaries, based on preliminary design, for the upper tug for the synchronous orbit mission are presented in Table 4-1.

Figure 4-5. Typical Battery Characteristics Required for Space Tug Missions



HIGH ENERGY DENSITY, SILVER-ZINC
PRIMARY LEM (DESCENT) BATTERY



VOLTAGE (20 CELLS)	28.0 VOLTS (MIN.)
CAPACITY	400 A.H. (MIN.)
	440 A.H. (TYPICAL)
DISCHARGE RATES	NOMINAL: 15.0 AMPERES
	MAXIMUM: 40.0 AMPERES

OPERATING CONDITIONS

TEMPERATURE	+ 40° F TO + 90° F
MODE OF OPERATION	VENTED
SHOCK	15 G's
VIBRATION	10 G's
LIFE	30 DAYS
WEIGHT	140 LBS. (MAX.)
VOLUME	1542 CUBIC INCHES
ENERGY DENSITY	94 WATT HOURS/LB. (TYPICAL)
	8.5 WATT HOURS/CU. IN. (TYPICAL)

(REF. G-8)

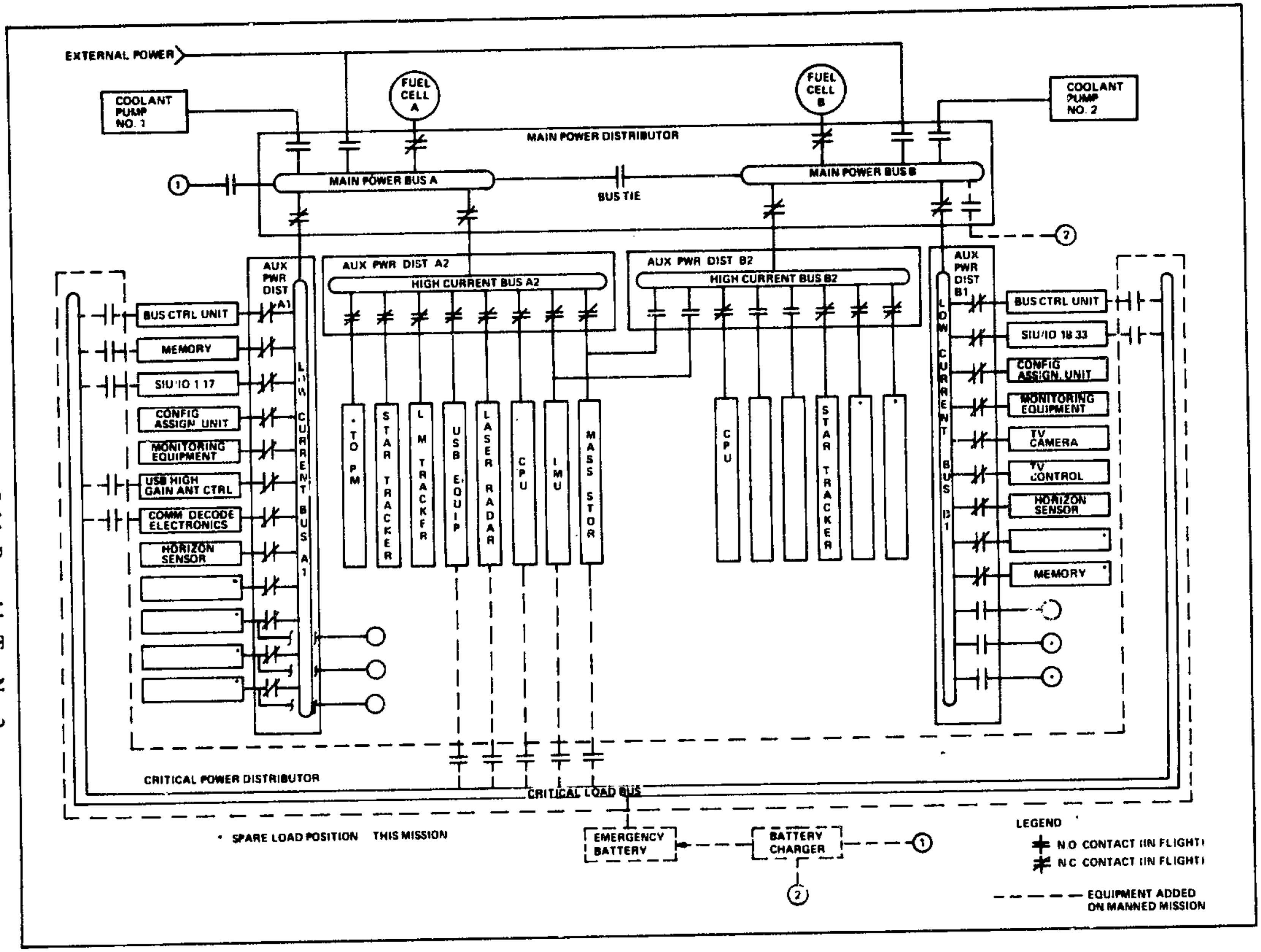


Figure 4-6. Power Distribution – Synchronous Orbit Reusable Tug No. 2

Table 4-1. Power Supply and Power Distribution Weight Summaries (Synchronous Orbit – Second Stage)

Component	Quan*	Wt (Ea)	Wt (Total)
Fuel Cell	2	169.0 lbs	338 lbs
Reactant (H_2)	–	5.5 lbs	11 lbs
Reactant (O_2)	–	44.5 lbs	89 lbs
H_2 Tank	2	7.0 lbs	14 lbs
O_2 Tank	2	14.0 lbs	28 lbs
DC Regulator	2	5.0 lbs	10 lbs
Total Power Supply Weight			490 lbs

* 100% redundancy of fuel cells tanks and reactants

Component	Quan*	Wt (Ea)	Wt (Total)
Power Distributor	1	39.0 lbs	39 lbs
Aux. Power Distributor	4	13.0 lbs	52 lbs
Wire and Cables	–	–	180 lbs
Mounting Hardware	–	–	20 lbs
Junction Boxes	8	2.0 lbs	16 lbs
Total Power Distribution Weight Summary			307 lbs
Total Power Weight Summary			797 lbs

4.2 FUEL CELL APPLICATION TO OTHER MISSIONS

4.2.1 Space Tug

4.2.1.1 Commonality of Power Requirements

As previously noted, with reference to Figure 3-1, the power/time requirements of all of the candidate space tug missions addressed in this study lie in the fuel cell domain. Also because of the commonality of mission operational phases (i.e., docking, separation, rendezvous) the trade factors which led to the recommendation of fuel cells for the synchronous mission apply equally well to the other candidate missions. The fuel cell system proposed has considerable flexibility in adapting to the three mission classes (expendable, manned reusable and unmanned reusable) by the modular addition of fuel cells, reactants, tankage and batteries to achieve the required mission reliability.

4.2.1.2 Power Weight Summary – All Tug Missions

Based on the 840 watt average power level derived from the synchronous orbit profile and active mission durations as follows, obtained by analysis of the mission profiles, a weight summary for the spectrum of space tug missions was prepared and appears in Table 4-2.

Space Tug Mission	Duration of Active Mission (hours)
Synchronous Orbit (First Tug), Reusable	18
Synchronous Orbit (Second Tug), Reusable	60
Synchronous Orbit, Expendable	24
Lunar Landing	29
Earth Orbital Operations	25
Lunar Orbital Operations	25
Planetary Mission, Reusable Tug	24
Planetary Mission, Exp. Tug	7
Four Stage Saturn V (see Note 1)	100

The following assumptions were made in computing the weights:

- Power distributor and auxiliary power distributor weights will be comparable to weights of present IU distributors.
- Flat cable techniques will be used resulting in an estimated total cable weight (signal and power) of 177 pounds.
- Reactant tankage weight is approximately 40% of reactant weight (vendor supplied rule of thumb). No consideration has been given to the selection of existing or actual cryogenic tanks for these missions.

4.2.1.3 Reusable Nuclear Shuttle (RNS) Earth/Moon Mission

The reusable nuclear shuttle mission presents a unique requirement to the astrionic module power system by virtue of the high peak power demands associated with operation of the NERVA engine. A preliminary analysis of RNS power requirements was made and the power profile, Figure 4-7, derived based on the following:

1. The RNS lunar mission timeline presented in Appendix A.
2. Power requirements during the three primary modes of NERVA engine operation are (Reference G12, G13):
    a. 9 KW start transient at cooldown initiation
    b. 5.2 KW required during engine burn
    c. 200 watts, average power during cooldown
    d. 60 watts, average power during coast

Note 1 – Sizing assumes that the other three Saturn V stages have separate power systems and are independent of the tug astrionic module, except for signal interface.

Table 4-2. Space Tug Power Weight Summary by Mission (All Weights in Pounds)

MISSION	TUG/CAT.*	KWH †	POWER ELEMENT									TOTAL EPS WEIGHT
			FUEL CELL		REACTANT	TANKAGE		POWER CONDITIONING		POWER †† DISTRIBUTION	BATTERY	
			QUAN	WEIGHT	WEIGHT	QUAN	WEIGHT	QUAN	WEIGHT	WEIGHT	WEIGHT	
SYNC. ORBIT (REUSABLE)	NO. 1/UM	7	2	338	13	4	5	2	10	307	–	673
	NO. 2/UM	50	2	338	100	4	40	2	10	307	–	795
SYNC ORBIT (EXPENDABLE)	NO. 1/UM	20	1	169	20	2	8	1	5	277	–	479
LUNAR LANDING	NO. 1/M	24	2	338	48	4	19	3	18	340	140	903
E.O. OPERATIONS	NO. 1/M	21	2	338	40	4	16	3	18	340	140	892
L.O. OPERATIONS	NO. 1/M	21	2	338	40	4	16	3	18	340	140	892
PLANETARY	NO. 1/UM R/U	20	2	338	40	4	16	2	10	307	–	711
	NO. 2/UM EXP.	6	1	169	9	2	4	1	5	277	–	464
FOUR STAGE SATURN V	NO. 1/UM	84	2	338	168	4	67	2	10	307	–	890

* - CATEGORY; UM-UNMANNED, M-MANNED
† - EXCLUDING ENERGY FOR STORAGE PHASES
†† - INCLUDES DISTRIBUTORS, CABLING (POWER & SIGNAL) & JUNCTION BOXES PLUS MOUNTING HARDWARE

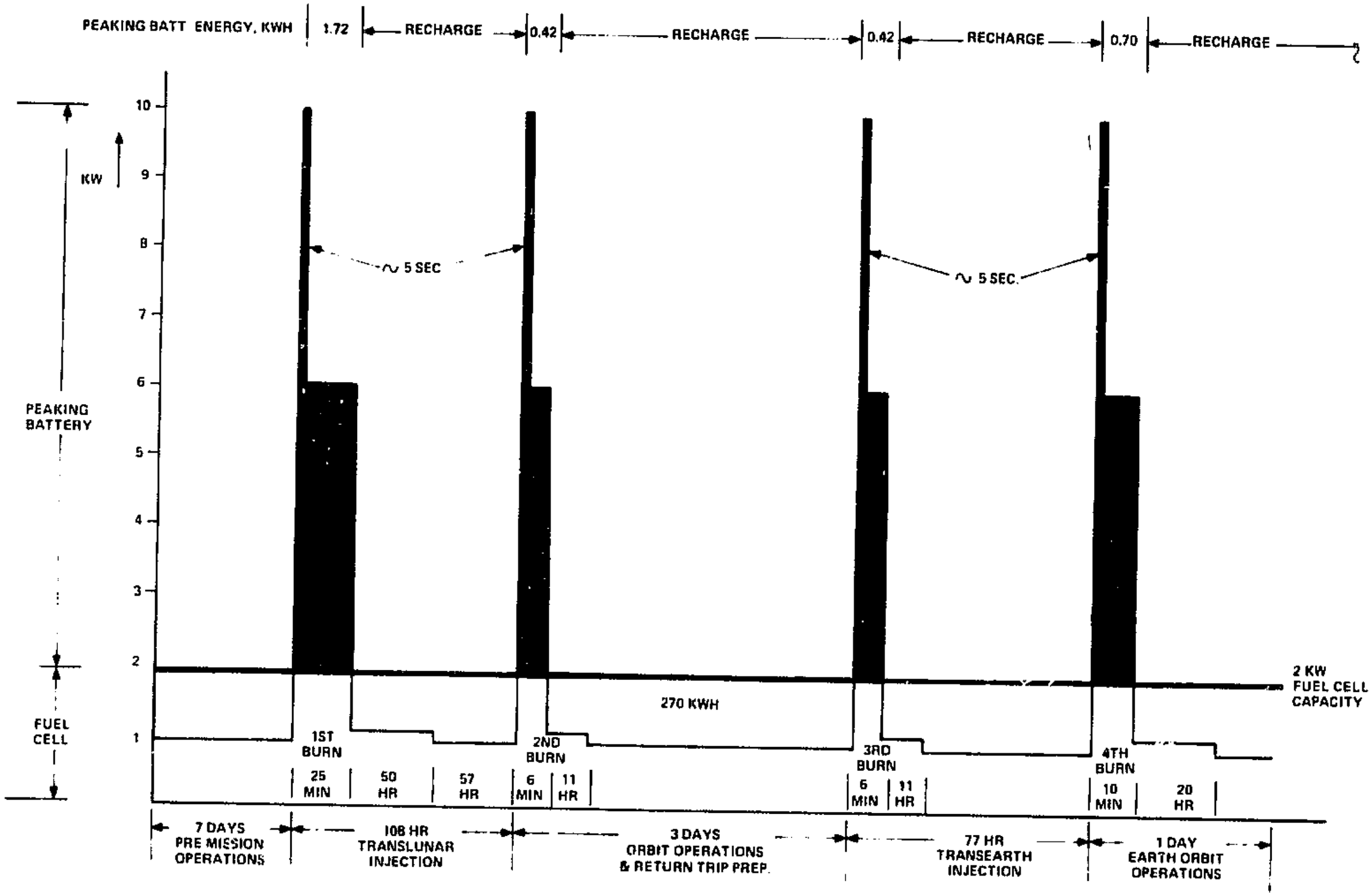


Figure 4-7. Fuel Cell and Peaking Battery Energy Requirements – RNS Mission

3. Electrical load imposed by astrionic equipment is 0.84 KW, average, in addition to the NERVA engine requirements.

4. No electrical loads, other than those itemized in (2) and (3), are fed by the astrionic module power system.

Examination of the power profile, Figure 4-7, shows that the 2 kilowatt fuel cell proposed for the other tug missions can adequately support the RNS mission, if peaking batteries are added to supply the short duration peak power demands imposed by NERVA (loads in excess of rated fuel cell capacity). The fuel cell system is particularly applicable to the RNS mission because of its total containment within the nuclear shuttle vehicle and resultant protection against NERVA engine radiation which could degrade externally mounted power sources, such as solar arrays.

4.2.1.3.1 Peaking Battery Requirements. As shown by the shaded areas of the power profile, the maximum energy requirement above the 2 KW fuel cell rating is 1.72 kilowatt-hours which occurs during the first NERVA burn. The secondary battery weight required to supply this energy, using silver-zinc secondary batteries at an energy density of 22 watt-hours per pound (55w-h/lb at 50% depth-of-discharge) is calculated as follows:

$$W_B = \frac{1.72 \times 10^3 \text{ watt-hrs}}{22 \text{ watt-hrs/lb}} = 78 \text{ lbs.}$$

The minimum time between engine burns is approximately 60 hours (2nd burn to 3rd burn), thus allowing ample time for battery charging between burns from the fuel cell primary source. Since all other burns are of considerably less energy content than the first, a battery sized for the initial burn will easily support the subsequent burns required for this mission.

4.2.1.3.2 Fuel Cell Energy Requirements. The unshaded portion of the power profile, Figure 4-7, represents energy to be supplied by the primary fuel cell power system. Summation of the power-time products from the profile yeilds a total energy requirement of 270 kilowatt-hours which must be supplied by the fuel cell system. At a specific reactant consumption of 0.83 lbs per kilowatt hour and with boiloff and contingency allowances of 10% each, the required weight of fuel cell reactant is:

W_R =	0.83 lb/kwh x 270 kwh	=	224 lbs
	plus 10% boiloff allow	-	23 lbs
	plus 10% contingency	-	23 lbs
	Total weight of reactant	=	270 lbs

This calculation allows for battery charging by the fuel cell system, at an energy efficiency of 70% for the silver-zinc batteries.

4.2.1.3.3 Preliminary Weight Statement. Based on the foregoing analysis, the estimated weight of the power system for the RNS earth/moon mission, with 100% fuel cell, reactant, tank and battery redundancy, is:

2 - 2 kw fuel cells @ 169 lb/each	–	338 lbs
2 - sets of peaking batteries @ 78 lb/set	–	156 lbs
270 lbs reactant/fuel cell x 2 F.C.'s	–	540 lbs
2 - sets reactant tanks @ 108 lb/set	–	216 lbs
*Power distribution and conditioning	–	307 lbs
Total weight of power system	–	1557 lbs

*From Table 4-2; weight for unmanned reusable missions

This weight appears compatible with values obtained for the other tug missions, leading to the conclusion that the combination of 2 KW fuel cells and silver-zinc secondary batteries, to support peak loads, constitute a prime candidate power system for the tug-supported RNS mission.

4.2.2 Other Space Transportation Systems

Reference has already been made to the current interest in fuel cell systems for space transportation systems such as the orbiter vehicle of the space shuttle. The Air Force has also indicated an intense interest in fuel cells as the primary source for hypersonic aircraft and the so-called "aerospace plane." A trade study conducted for the Air Force entitled "Fuel Cells for Air Force Requirements" (Reference G-9) in October 1967 concludes that fuel cells are the best choice for manned orbital and space missions which extend for more than a few days and involve vehicles designed primarily for transportation purposes. This conclusion is supported by trade-off data from that study which is presented as Table 4-3, showing that fuel cells represent a superior choice in terms of low weight and low cost per mission based on projected performance in the next 20-30 years. It should be noted that total power system cost for the aerospace plane program are distributed over 50 missions to obtain the per-mission weight and cost figures presented in Table 4-3. The significant weight and cost increases for the "single-shot" sources (silver-zinc primary and metal-oxygen batteries) are due to the need for replacement after each mission, whereas, the fuel cells, secondary batteries and turbine-generators are reusable sources. Silver-zinc secondary battery weight is high because of the conservative depth of discharge (40%) and specific energies 40 WH/LB employed.

4.3 CONSIDERATIONS FOR INTEGRATED TUG POWER SYSTEM

One goal of the study is to make a "quick look" assessment of the advantages and disadvantages of an integrated tug power system, that is, a central power system located in the astrionic module and capable of supplying the total combined mission requirements of the astrionic loads, crew module loads and propulsion module loads. The obvious advantages of this approach are reduced power system design time and reduced weight through elimination of duplicate power sources. Potential disadvantages include overdesign of the power system for missions on which the crew module is not flown and total dependence of the crew module on the astrionic module for its electrical power supply (except for portable life support systems). The use of an integrated power system would parallel the Apollo command module/service module concept in which all primary power sources (fuel cells) for the crew support functions are located in the adjacent service module.

Table 4-3. Weight and Cost Trade-Offs for Aerospace Plane Mission*

Power Source	Wt-Cost per Mission**	
	Wt (lbs)	Cost ($)
High Current Density Fuel Cell at 0.9 Volts/Cell		
Fuel Cell Battery (15 lbs/KW)	15.0	10.00
Cryogenic H_2 + Tankage (9.2 lbs H_2 x 2.5)	23.0	4.80***
Cryogenic O_2 + Tankage (73 lbs O_2 x 1.5)	110.0	10.40***
Total Fuel Cell Battery System	148.0	25.20
Low Current Density Fuel Cell at 1.1 Volts/Cell		
Fuel Cell Battery (25 lbs/KW)	25.0	8.10
Cryogenic H_2 + Tankage (7.5 lbs H_2 x 2.5)	18.8	3.76***
Cryogenic O_2 + Tankage (59.8 lbs O_2 x 1.5)	89.7	8.40***
Total Fuel Cell Battery System	133.5	20.26
Silver-Zinc Secondary Battery (cycle life of 500 at 40% discharge, 40 watt-hour per lb of battery, or 25 lbs/KWH)	2,500.0	100.00
Single-Shot Silver-Zinc Primary Battery (100 watt-hour per lb of battery, or 10 lbs/KWH)	1,000.0	20,000.00
Possible Single-Shot Metal-Oxygen Battery (200 watt-hour per lb of battery, or 5 lbs/KWH)	500.0	40,000.00
Future Gas Turbine-Generator Set (H_2–O_2)		
Gas Turbine–Generator Set Equipment	20.0	1.0
Cryogenic H_2 + Tankage (19 lbs H_2 x 2.5)	47.5	9.50***
Cryogenic O_2 + Tankage (151 lbs O_2 x 1.5)	226.5	21.10***
Total Gas Turbine–Generator	294.0	31.60

* Mission Time 100 Hours for 50 Missions; Auxiliary Power = 1 KW and 100 KWH
** For Target or Projected Performance in 20 – 30 Years
*** Assumed Cryogenic Tanks at $10.00 per lb of Dry Tank and Last (Life) for all Missions (50)

4.3.1 Crew Module Requirements

Although study time limitations precluded a detailed analysis of the electrical loads imposed on the tug power system by the crew module, preliminary estimates of these loads were obtained from the crew module study contractor. These estimates are presented in Table 4-4.

Table 4-4. Power Requirements for Crew Module

Crew Module Function	Power (watts) (3 man crew)	Power (watts) (6 man crew)
Crew and Crew Support	15	30
Food Management	20	40
Water Management	77	128
Temp. & Humidity Control	110	184
Atmospheric Purification	230	384
Instrumentation & Controls	110	110
Atmospheric Supply	250	418
Unscheduled Experiments	0	50
Subtotals	812	1344
20% Contingency	162	269
Subtotals	974	1613
Displays (astrionic equipment)	840	840
	1814	2453

4.3.2 Propulsion Module Requirements

No estimate of tug propulsion module loads was available for this study. However, it is estimated that these loads would be comparable to those estimated for the space shuttle orbiter which are approximately 500 watts (see Reference G-10). Duration of propulsion loads are expected to be in the order of minutes and, thus, would be handled by the addition of peaking batteries in preference to increasing prime power source rating.

4.3.3 Impact on Power System Sizing

Assuming that peaking batteries can, in fact, be used to support propulsion module loads the effect of an integrated power system concept would be to increase the required rating of the individual fuel cell modules and to double or triple the amount of fuel cell reactant to be carried at time of tug activation. The total load requirements obtained by adding the requirements for the 3 and 6 man crew modules to the 0.84 KW average load for astrionics equipment are 2.65 KW and 3.29 KW, respectively. Thus, an increase to a 3 KW basic fuel cell module would probably be required unless the crew module load duty cycles are significantly less than 100%. If a more detailed analysis of crew module loads reveals that substantial load blocks are intermittent in nature it is possible that the 2 KW units could be retained, as this unit has an overload capability of 4.5 KW for a 5 minute period.

In any case, the amount of additional fuel cell reactants required would be significant. Based on the 60 hour mission time of the synchronous orbit second tug and the crew module power requirements presented in 4.3.1, the additional reactant and tank weight for crew module support would be:

3 man crew – 253 lbs

6 man crew – 342 lbs

These figures are based on 100% fuel cell redundancy and a reactant consumption rate of 0.83 lbs/KWH.

Required weight of peaking batteries cannot be estimated until a thorough analysis of the total load profile for the integrated power system is made including the magnitude and duration of propulsion module peak load requirements.

4.4 LUNAR LANDING MISSION

A secondary mission addressed in the power system analysis is the lunar landing mission for the space tug. From a power system standpoint this mission is characterized by full power operation for time intervals measured in hours followed by very low power operation for periods in excess of one month (lunar and orbital storage). Unique constraints imposed on the power system by this particular mission are:

- Operation in extended periods of sunlight and darkness on lunar surface.
- Non-interference with vehicle maneuvering capability as required in lunar take-off and landing.
- No exposure of power system components susceptible to damage by the lunar environment.

4.4.1 Mission Power Requirements and Source Selection Criteria

Due to non-availability of definitive electrical load requirements for the tug astrionic equipment, the following load figures were estimated from present Instrument Unit requirements and on the premise that a passive (non-power consuming) thermal control would be used:

Equipment Type	Flight Phases	Storage Phases
Guidance and Navigation	900 watts	--
Instrument & Communications	600 watts	100 watts
Battery Charging	--	100 watts
Totals	1500 watts	200 watts

Using these power requirements a preliminary power profile and energy budget for the lunar landing mission were developed, as shown in Figure 4-8.

Selection of a power source to satisfy the mission requirements identified in Figure 4-8 is based on the following criteria:

- Weight - compatible with total astrionic module weight allowance
- Volume - compatible with total astrionic module volume allowance (storable for launch in cargo bay of shuttle)
- Cost Effectiveness - optimized for both reusable and expendable versions of space tug
- Reliability - adequate for manned missions but suitable for remote operation to allow use on unmanned versions
- Mission Compatibility - minimum interference with primary space tug mission objectives including docking operations, long term orbital storage and lunar landing and take-off

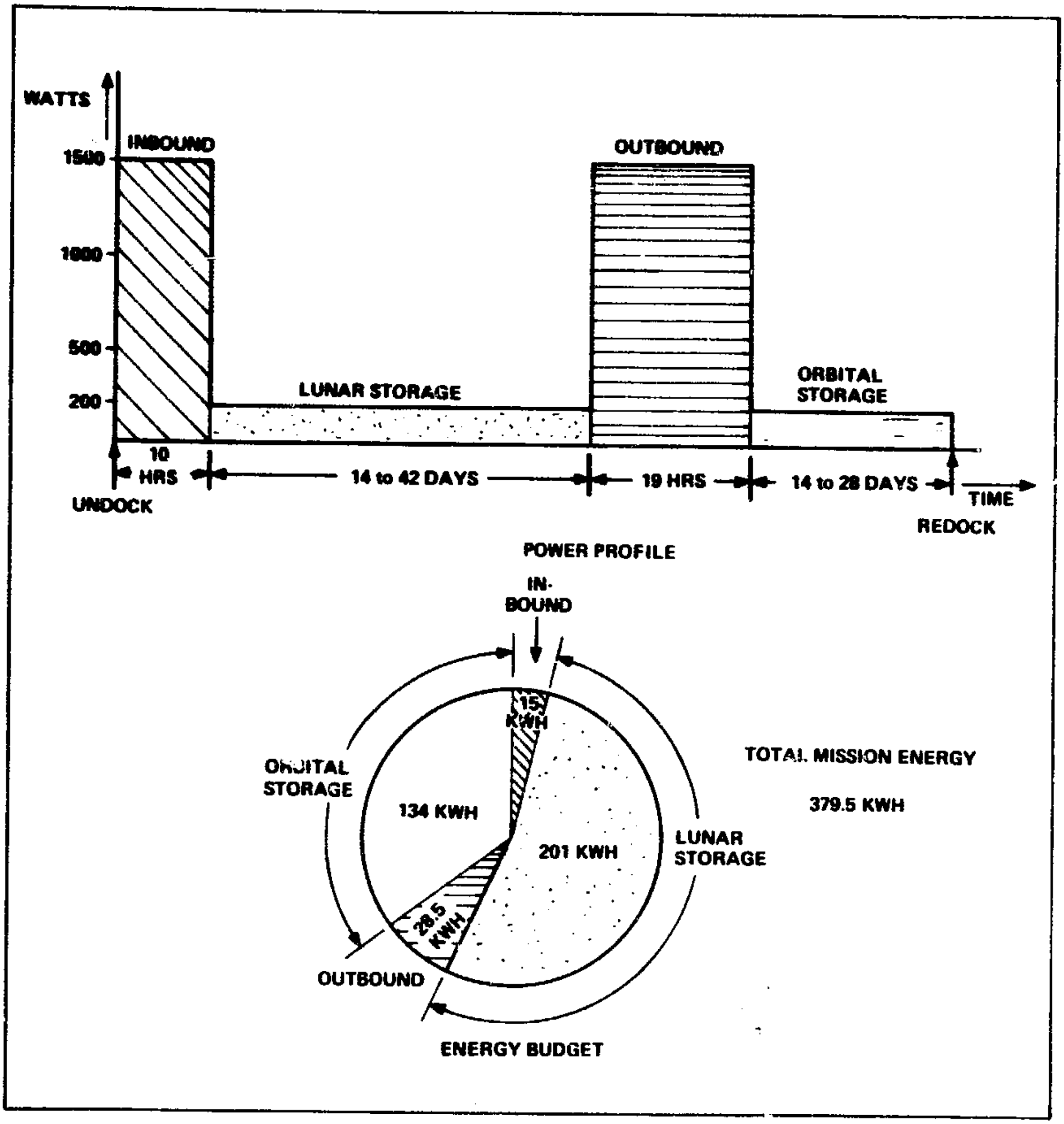


Figure 4-8. Power/Energy Requirements Lunar Landing Mission

- Autonomy – independent of other space tug modules and other space vehicles except for resupply and refurbishment from space station

- Crew Safety – maximized by use of proven technologies and with adequate provisions for emergency operating modes

4.4.2 Applicability of Power Source Technologies

4.4.2.1 Solar Array/Battery System

A solar array/battery power system could satisfy all of the stated criteria for the lunar landing mission with the important exception of mission compatibility. The two major areas of incompatibility are:

- The need for array orientation which might conflict with tug maneuverability and the presence of external appendages which would cause interference with docking operations, EVA and communications antenna patterns.

- The long period of darkness which would require an unacceptable weight of secondary batteries to power the system during lunar night (14 earth-day duration).

Solar array systems would also be less cost effective than chemical conversion systems as shown in the cost comparison chart, Figure 4-9 (Reference G-2).

4.4.2.2 Primary Battery System

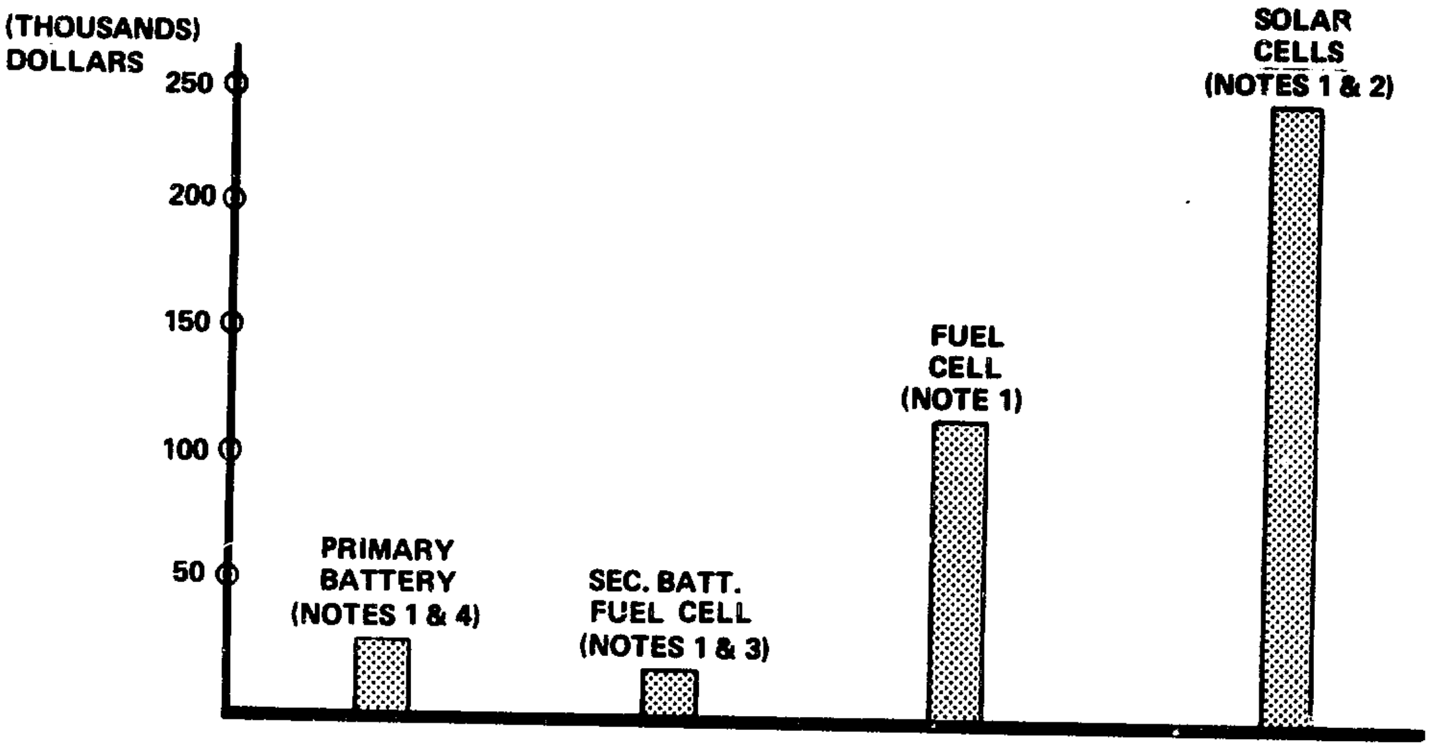
A primary (non-rechargeable) battery system is unacceptable for the tug lunar landing mission due to excessive weight. Using the projected energy density achievable with 1973 technology of 100 watt-hours per pound, the weight of primary batteries alone (excluding other components of the power and distribution system) would be 2500 pounds.

4.4.2.3 Fuel Cell System

A fuel cell system satisfies all of the power source selection criteria but is not the most cost effective system when applied to the high power, short time duration flight phases. This is due to the fact that the fuel cell must be sized to support the maximum power demand (1.5 KW) which, in the case of the lunar landing mission, represents only a small percentage of the total mission time. It does, however, represent the lightest applicable power source for this mission as shown in the weight comparison chart, Figure 4-10 (Reference G-2).

4.4.2.4 Secondary Battery/Fuel Cell System

A system comprising secondary batteries to support the high power short duration phases of the lunar landing mission in conjunction with small fuel cell units capable of recharging the batteries during the extended lunar and orbital storage periods offers an attractive power option for the lunar landing mission. It appears to be the most cost effective system because the lower cost batteries can readily supply the mission flight phases (19 hours maximum duration) with a resultant drastic reduction in required kw rating and, hence, in cost of the fuel cells. While considerably heavier than the fuel cell system (Figure 4-10) it is more readily expandable with increasing astrionic load requirements by virtue of the simplicity of a battery system (e.g., peripheral equipment such as reactant tanks, water recovery systems, etc. needed for large fuel cells are not required).

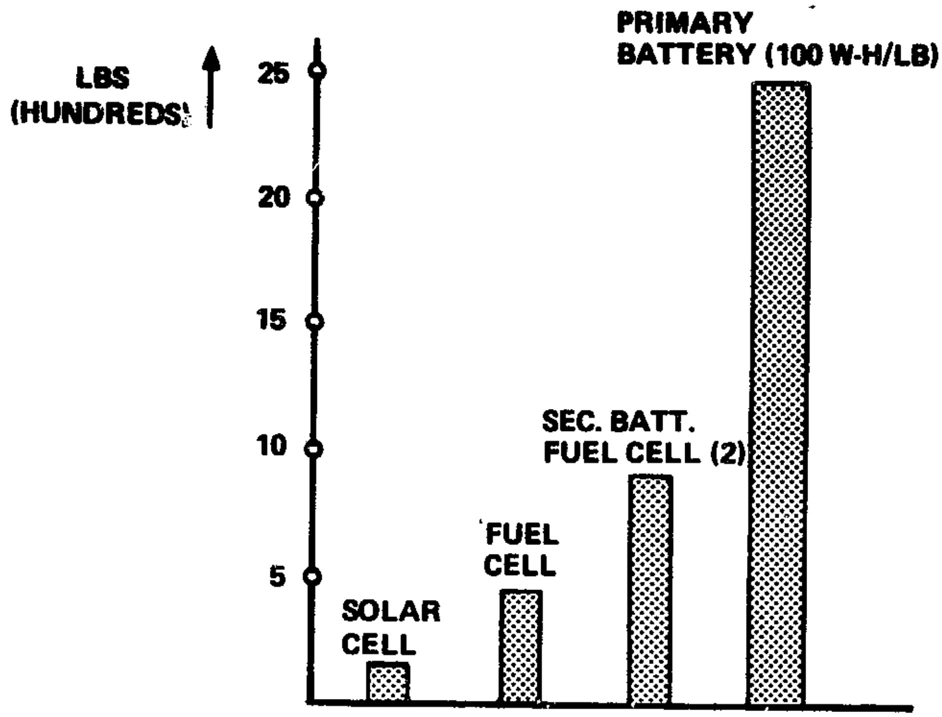


(REF. G-2)

NOTES:

1. POWER SOURCE ONLY – COST OF POWER CONDITIONING & DISTRIBUTION EQUIPMENT NOT INCLUDED.
2. BASED ON SILICON, PANEL MOUNTED CELLS ORIENTED TO ± 15 ° WITH NICKEL CADMIUM BATTERY ENERGY STORAGE.
3. BASED ON SECONDARY BATTERIES FOR FLIGHT PHASES (1.5 KW for 24 HRS., MAX.; 200 WATT FUEL CELLS FOR LUNAR STORAGE PHASE (720 HOURS, MAX.).
4. BASED ON 1.5 KW FOR 24 HRS., 100W FOR 720 HOURS.

Figure 4-9. Power Source Cost Comparison



(REF. G-2)

NOTES:

1. POWER SOURCE ONLY - WEIGHT OF POWER CONDITIONING & DISTRIBUTION EQUIPMENT NOT INCLUDED.
2. BASED ON USE OF SECONDARY BATTERIES FOR LUNAR DESCENT & ASCENT PHASES; FUEL CELLS FOR LUNAR STORAGE PHASE.
3. BASED ON 1.5 KW ASTRIONICS LOAD IN FLIGHT PHASES.

Figure 4-10. Power Source Weight Comparison

4.4.2.5 Nuclear Power Systems

The lunar landing mission does not represent a feasible application of nuclear power systems, primarily because of the high cost and shielding weight penalties involved with such systems. In addition, the development status of both radioisotope and reactor heated power systems makes their availability for the projected space tug hardware procurement dates very questionable.

4.4.3 Conclusions

Preliminary analysis of the electrical power requirements for the tug lunar landing mission and a survey of available power conversion technologies indicate that these requirements can be supported by a secondary battery/fuel cell system at minimum cost. This system offers a distinct advantage is cost effectiveness due to the use of batteries for the high power, short duration, orbit to lunar surface and return phases. A system comprising fuel cells as the primary source with supplemental peaking batteries is significantly lighter but is more complex and costly than the hybrid battery/fuel cell system.

4.4.4 Preliminary Design – Lunar Landing Mission Power Option

4.4.4.1 Battery Sizing

Reference to Figure 4-7 shows that the maximum energy which must be supplied by the battery system is 28,500 watt-hours during the lunar ascent phase of the mission.

For silver-zinc secondary batteries, which provide the best obtainable energy density (watt-hours per pound), the maximum permissible depth of discharge to achieve the desired cycle life (72 charge-discharge cycles in 3 year, 36 mission tug lifetime) is 45% without voltage degradation.

The energy capacity of the battery system must then be:

$$\frac{28{,}500 \text{ watt-hrs.}}{0.45} = 63{,}400 \text{ watt-hrs.}$$

Assuming an energy density of 55 watt-hours per pound, predicted achievable by Ag-Zn secondary batteries by 1975, the total weight of batteries required is:

$$\frac{63{,}400 \text{ watt-hours}}{55 \text{ w-h/lb.}} = 1150 \text{ lbs.}$$

Based on a 28 volt DC system, the total battery ampere-hour requirement is:

$$\frac{63{,}400 \text{ watt-hrs.}}{28 \text{ volts}} = 2265 \text{ ampere-hours}$$

Assuming that the total ampere hour requirement is supplied by ten (10) 225 amp-hr units, failure of a single battery would result in loss of only one-tenth of the system capacity.

4.4.4.2 Battery Charging

Using a 70% watt-hour efficiency, which is typical for silver-zinc secondary batteries, the energy which must be replaced during the lunar storage charging period is based on the battery energy expended during the ten (10) hour lunar landing phase or: 10 hrs. x 1500 watts = 15,000 watt-hours, plus the charging losses. Charging energy which must be supplied by the fuel cells is then:

$$\frac{15{,}000 \text{ watt-hours}}{0.70} = 21{,}400 \text{ watt-hours}$$

Based on a 30 hour charge rate for the batteries, the required charging power is:

$$\frac{21{,}400 \text{ watt-hours}}{30 \text{ hours}} = 715 \text{ watts}$$

To minimize required fuel cell capacity and, hence, cost and because there is ample time to accomplish battery charging in serial fashion, it is assumed that one-tenth of the total charging power will be supplied to a single battery until it reaches the charged state at which time charging power will be switched to the second unit, etc., until all units are fully recharged, in a total elapsed time of 300 hours (12.5 days). Any additional available charging time will be used to sequentially scan and trickle charge the batteries to replace any energy lost due to self-discharge and insure full battery capacity for the ascent phase.

4.4.4.3 Fuel Cell System

The fuel cell system to perform the battery charging function and to supply the power to astrionic loads which are active during the lunar storage phase consists of a single 200-watt unit similar to the Allis-Chalmers radiation cooled unit developed for Air Force orbital operations, with its associated reactant tanks and electrical supervisory subsystem. Due to its radiation cooling concept which utilizes louvered shutters over a variable emissivity heat rejection surface, its light weight (31 lbs.) and small size, this type unit is greatly simplified compared to the liquid cooled units flown on past missions. The fuel cell approaches a battery in installation simplicity since the only peripheral equipments required are the hydrogen and oxygen reactant tanks and ports for exhaust of product water. The physical configuration of the proposed fuel cell is shown in Figure 4-11.

4.4.4.4 Schematic

The power distribution scheme for the lunar landing mission is illustrated in Figure 4-12. A segmented distribution bus is shown with provisions for 100% redundancy of the battery charging system shown by dashed lines. In this concept the astrionic loads needed for the storage phases of the mission would be switched to the "storage bus," removing load from the A and B battery buses during battery charging. The bus tie switches between the storage bus and the A and B battery buses would be closed, during emergency modes only, to allow powering critical astrionic loads from the fuel cell(s) in the event of battery system failure.

4.4.4.5 Weight Summary

Table 4-5 is a weight summary of the baseline and redundant versions of the battery/fuel cell system. Included in the table is a summary of the alternate system using two kilowatt fuel cells as the primary source, with peaking batteries provided for loads in excess of fuel cell capacity and for emergency power. As shown in this summary, a weight saving of 648 pounds is obtainable with the primary fuel cell system, without redundancy.

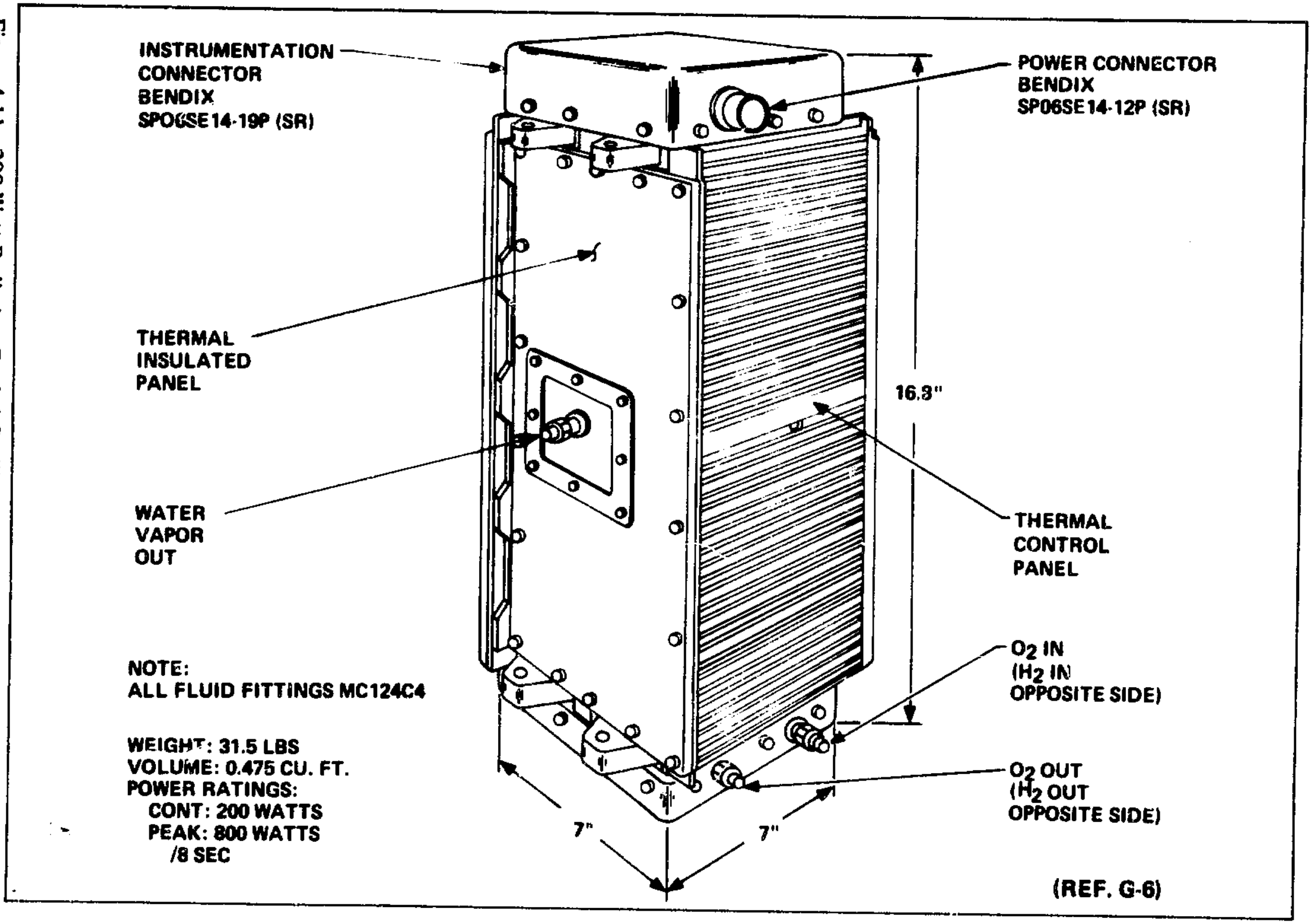


Figure 4-11. 200 Watt Radiation Cooled Capillary Matrix Fuel Cell

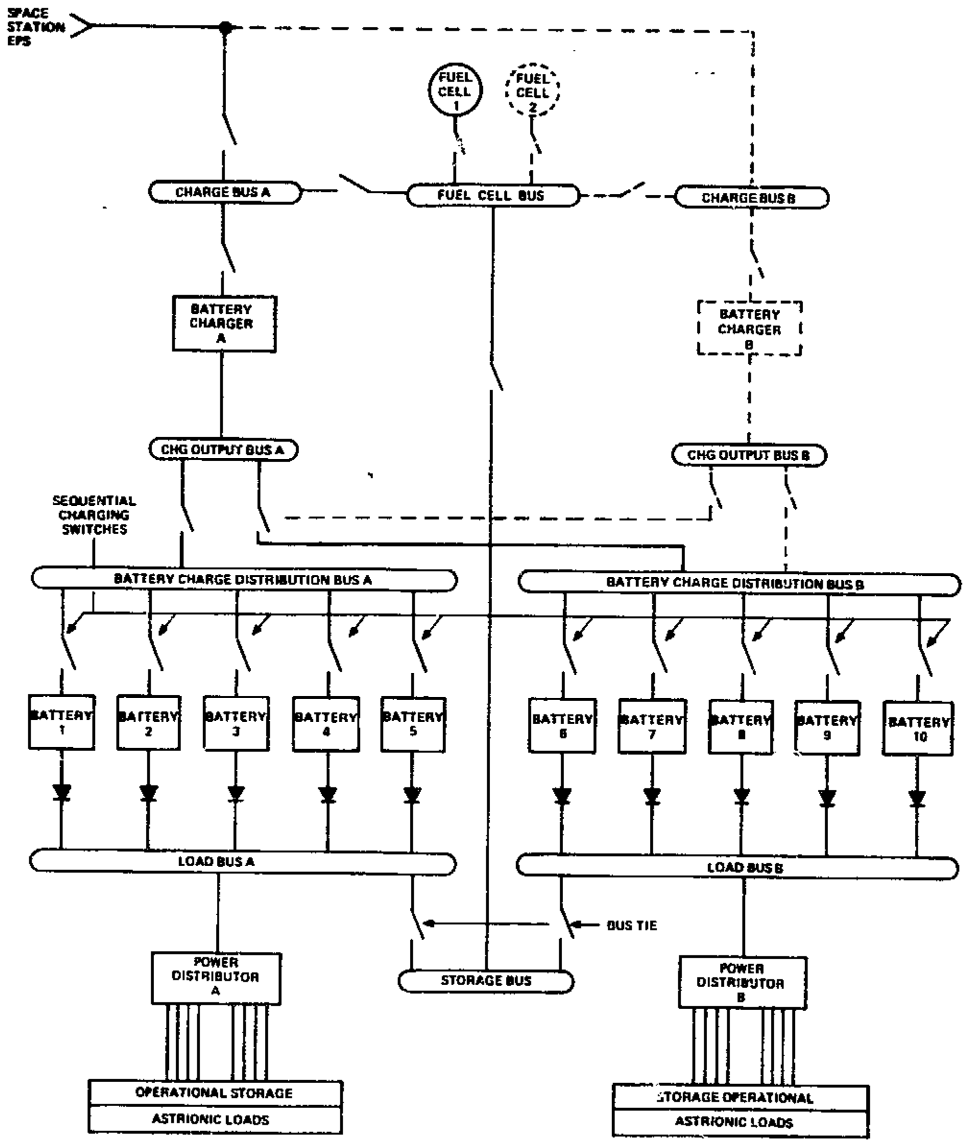


Figure 4-12. Power Distribution – Lunar Landing Mission Alternate Option

Table 4-5. Weight Summary

System	Secondary Battery/Fuel Cell				
		Non-Redundant		Redundant	
Component	Weight Each (Lbs)	Quan	Weight Total (Lbs)	Quan	Weight Total (Lbs)
Battery	115	10	1150	10	1150
Fuel Cell	31	1	31	2	62
Reactants (H_2, O_2)	–	–	119	–	238
Reactant Tanks	–	2	48	4	96
Battery Charger	8	1	8	2	16
DC Regulators	15	2	30	4	60
Wire & Cable *	–	–	177	–	250
Distributors	35	5	175	5	175
Relays & Switches	TBD	TBD	30	TBD	50
Mounting Hardware	TBD	TBD	15	TBD	20
Total EPS Weight			1783		2117

*Cable weight extrapolated from IU cable weight of 482 lbs. assumes use of flat cable techniques for all wires AWG No. 16 and smaller (50% weight reduction) and reduction in signal wiring by virtue of Data Bus (50% quantity reduction).

4.5 APPLICATION OF POWER MANAGEMENT CONCEPTS

Although this study did not consider the application of computerized power system management, the concept has potential benefits which should be investigated in future tug power studies. Advantages which could be realized through power management by the on-board computer and data management subsystems include:

- Relieving the crew of routine supervision of the power system
- Electrical load budgeting (flattening of the power profile) for optimum utilization of energy and power resources
- Load priority assignment and update
- Independence from ground commands at long communication distances (of particular significance to unmanned planetary missions)
- Increased EPS reliability and crew safety through continuous computer monitoring, crew display and high speed reaction to abnormal system conditions (e.g., switchover to standby or emergency units)

Automated power management is receiving increasing attention as power system complexity increases and is being analyzed in depth for the space shuttle and other future space transportation systems. The concepts evolved in these studies should be reviewed for applicability to the space tug which has comparable reliability requirements and mission durations.

4.6 FUTURE POWER SYSTEM TECHNOLOGY

Technical advances in all of the power technologies including solar cells, batteries, fuel cells, nuclear systems, both isotope and reactor heated, with their many conversion systems (Brayton and Rankine cycle, thermoelectric and thermionic) are too numerous to discuss within the scope of this report. A brief synopsis of the power level and era of application is presented in the Space Power System Forecast, Figure 4-13 (Reference G-11).

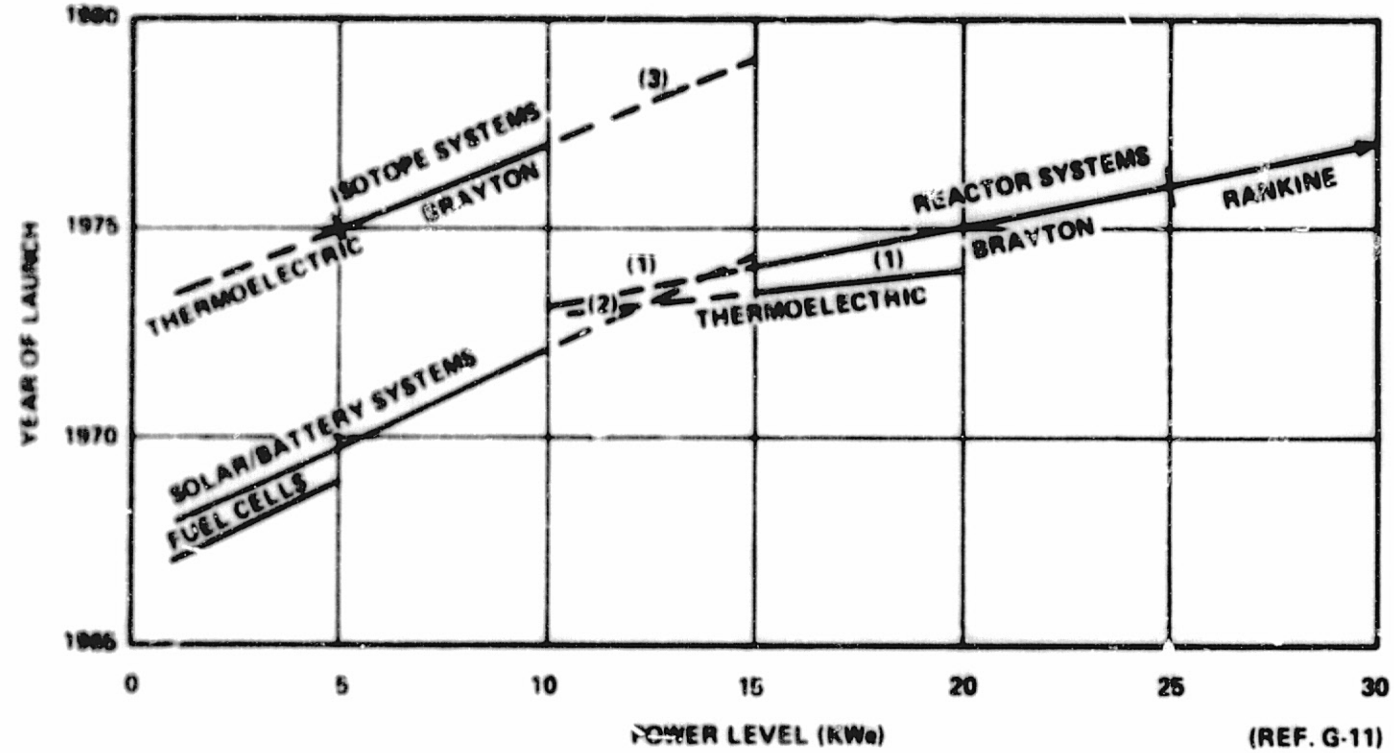
Since it is the conclusion of this report that fuel cells offer the most advantage to the space tug and other space transportation systems it is more relevant to cite some of the advances in this specific technology. The areas receiving most development attention according to the Power Information Center (PIC) briefs which are distributed on a monthly basis by the Interagency Advanced Power Group are:

- Improved Fuel Cell Catalysts
- Regenerative Fuel Cells
- High Current Density Fuel Cells
- Improvement of Electrode Structures
- Zinc-Oxygen Batteries (semi-fuel cells)
- Thin Film Fuel Cells
- Evaluation of New Fuel Cell Concepts

Basically these efforts are directed at the following objectives relative to space power applications:

- Reduction in fuel cell cost
- Reduction in weight
- Extension of operating life
- Reduction in complexity
- Improved methods of reactant storage

Since weight and cost are of primary concern to the space transportation system applications the predictions of Glasser and Fleischer included in the previously referenced study "Fuel Cells for Air Force Requirements" (Reference G-9) are included as Figures 4-14 and 4-15 to show anticipated reductions in specific weight and cost, respectively. Note that the costs presented are costs of fuel cell materials, only, and do not reflect engineering or development costs which will vary widely as a function of quantities produced for a given space application.



NOTES:

1. POWER LEVELS SHOWN ARE AVERAGES AT THE LOAD BUS. THESE ARE APPROXIMATELY 50% OF THE PEAK-LOAD POWER DEMANDS.

2. BATTERIES ARE USED FOR ALL SYSTEMS FOR LOAD EQUALIZATION.

3. DASHED LINES SHOW REGIONS OF APPLICATION WITH WEIGHT(1) COST(2) OR INVENTORY(3), PENALTIES.

Figure 4-13 Space Power-System Forecast

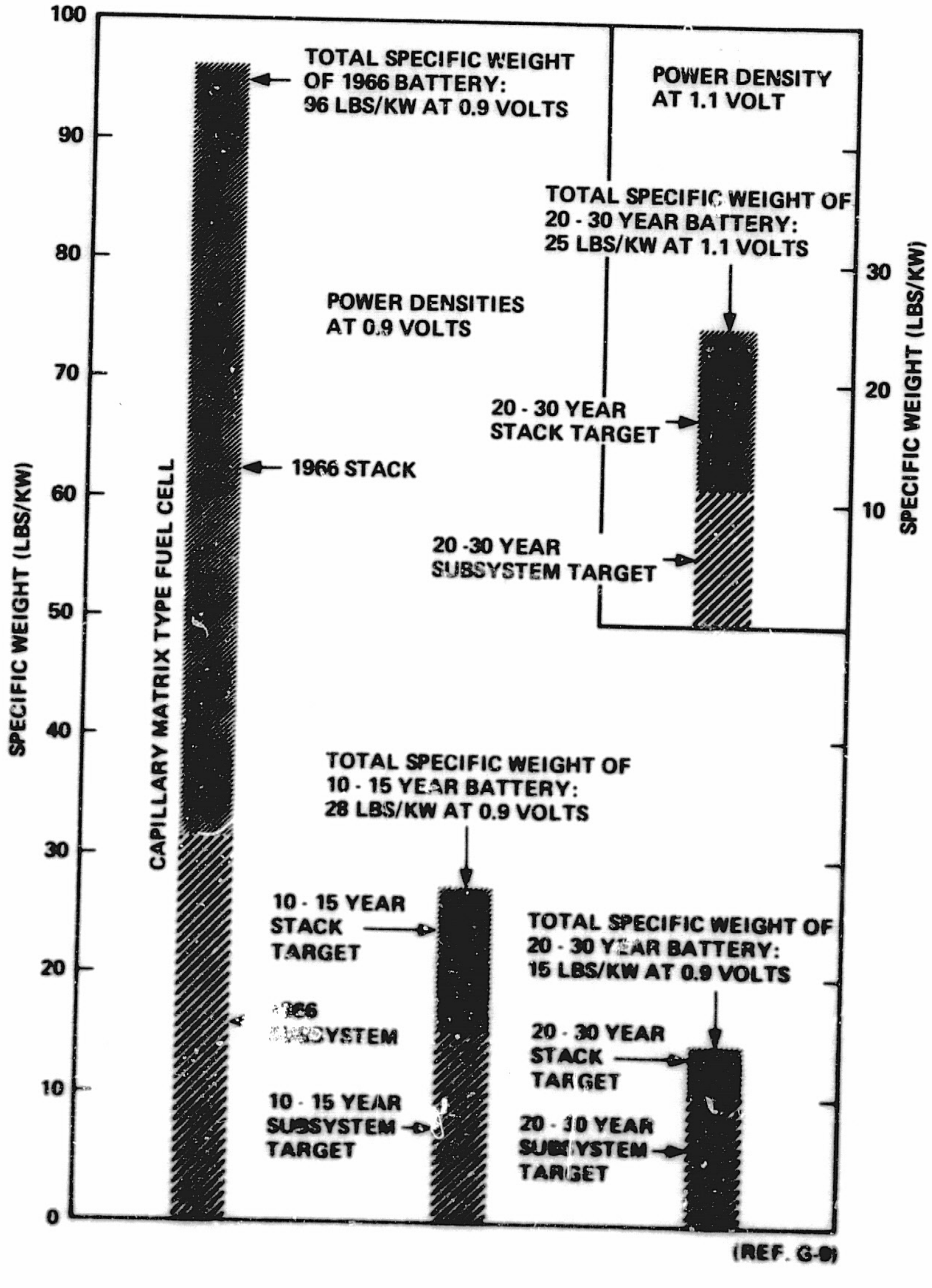


Figure 4-14 Projections of Power Density or Specific Weight for H_2-O_2 Fuel Cell Battery

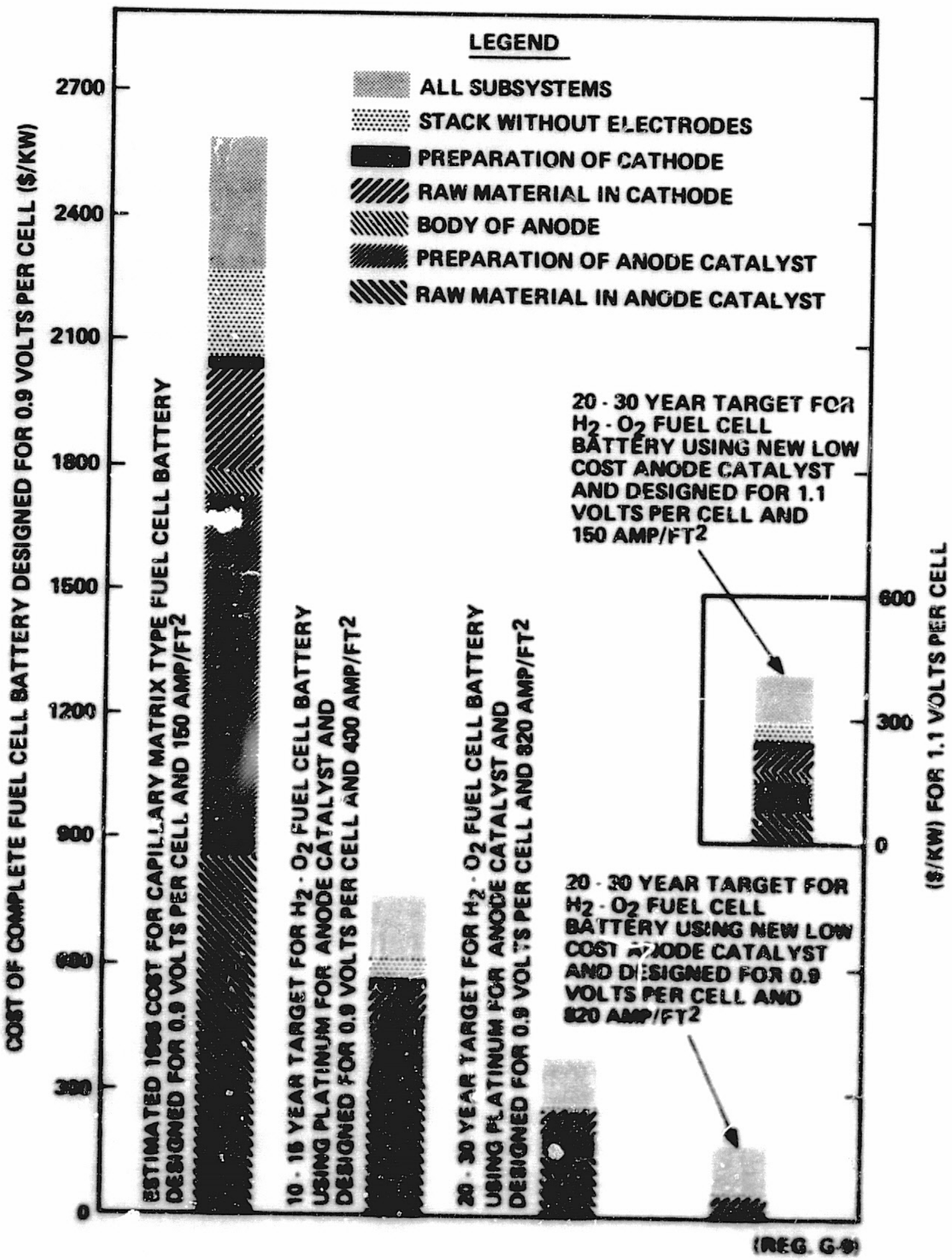


Figure 4-15. Estimated Costs of H_2-O_2 Fuel Cell Batteries

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*Intersociety Energy Conversion Engineering Conference

APPENDIX H

SPACE TUG COMMUNICATIONS ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

The purpose of this appendix is to present guidelines, assumptions, considerations, and a preliminary design for space tug communications. The objectives of this study were to:

- Identify the parameters which affect the design of the tug communications
- Configure composite tug communications which would cover the spectrum of tug operations
- Identify the functional components required for each tug mission
- Estimate the power, size, and weight of the communications components to support the overall astrionic module design

Of the basic tug design missions for astrionics as described in a previous appendix of this report, the unmanned synchronous mission received the most emphasis. However, all of the missions were considered sufficiently to permit a composite design to satisfy the requirements of all known missions.

The level to which the communications could be conceptually designed was limited by the lack of definite communications requirements in the form of bandwidth, data rates, redundancy required, etc. However, sufficient groundrules and guidelines did exist to make the conceptual design meaningful for planning purposes.

2.0 GUIDELINES, ASSUMPTIONS, AND CONSTRAINTS

2.1 GENERAL

The overall requirement for tug communications is simply to provide the communication services required for successful tug operations. These services will include uplink command and voice channels, provision for tracking and ranging, and downlink data, voice, and TV on Unified S-Band (USB). VHF ranging and EVA communications will be provided as required

2.2 GUIDELINES

The general guidelines for the space tug astrionic system study are presented in a previous appendix. Those which most directly affect the communications layout are

- Space maintenance. Replacement of replaceable units in space
- Compatibility with shuttle, earth orbiting space station (EOSS), lunar orbiting space station (LOSS), reusable nuclear shuttle (RNS), and other tugs
- Compatibility with either Manned Space Flight Network (MSFN), Deep Space Network (DSN) or Air Force Space Ground Link System (SGLS)
- Compatibility with a data relay satellite system (DRSS) or commercial communications satellite

- Capable of maintaining a quiescent status for up to 180 days.
- Minimize power and weight.

2.3 ASSUMPTIONS

The following assumptions have been made in conjunction with this study:

- The prime communications link with the tug will be in the S-band frequency range of 2.1 to 2.3 GHz.
- The manned missions will require VHF voice for communications with other elements such as shuttle, space stations, and RNS. The VHF system can also provide ranging data. EVA communications will be via VHF. The storage mode command receiver will also operate in the VHF band. Two VHF bands (136 - 150 MHz and 260 - 297 MHz) are available for the above.
- DRSS and/or Tactical Communications Satellite (TACSAT) will be operating in 1977 when the space tug is scheduled to begin operating.
- Any support for the tug payload (other than intercom service or TV) required of the communications subsystem will be through the data bus standard interface units (SIU).
- The DSN ground stations (85-feet and 210-feet diameter antennas) will be available for lunar and planetary missions. The 85-foot antenna will be available when TV is required from the tug at synchronous orbit altitude.

2.4 CONSTRAINTS

The following are considered constraints which will affect the communications design:

- The vehicle diameter of 14 feet will require omni antennas to be placed on opposite sides to obtain near omni directional coverage.
- Astrionic module height of 4 feet places a constraint on the mounting of the 4 feet diameter hi-gain antenna if it is located in the astrionic module. The maximum allowable module diameter also means the hi-gain antenna must be stored within the vehicle and erected for use.
- Compatibility with MSFN and AF SGLS may require two sets of equipment depending on the mission, since MSFN and SGLS Unified S-band equipment is not presently compatible in all aspects.
- Space maintenance by replacement of replaceable units will place some constraints on the electrical and mechanical design of these units.
- The requirement to transmit commercial quality TV from the tug is the most severe constraint for communications. This requires the use of a steerable antenna on the tug and use of the 85-foot antenna on the ground for the synchronous orbit mission.

2.5 MISSIONS

The detailed mission descriptions are given in a previous appendix of this report. The unmanned synchronous orbit (reusable or expendable) mission was used as the baseline for this study. This communications study considered the following tug missions:

- Synchronous orbit; unmanned, reusable or expendable
- Lunar landing; manned with unmanned capability
- Low earth orbit; manned with unmanned capability
- Lunar orbit; manned with unmanned capability
- Planetary; unmanned. This is a boost mission only and maximum range was given as 30,000 NM.
- Reusable nuclear shuttle; manned or unmanned. This mission involves the use of the tug astrionic module equipment for the RNS.
- Four Stage Saturn V; manned or unmanned. The tug is the 4th stage of a Saturn V launch vehicle and, in addition to providing the launch functions, the astrionic module may operate on any mission up to lunar distances.

3.0 SUMMARY OF RESULTS

The results of this study are:

- An estimation of the tug communication requirements
- A preliminary design for space tug communications
- The identification of the equipment required to meet communications requirements, estimation of equipment characteristics and mission usage
- Identification of the design parameters and considerations

A summary of tug communications interface requirements is given in Table 3-1 and illustrated in Figure 3-1. The most severe requirement is that for TV transmission from the tug. This requirement will dictate a high-gain antenna on the tug and the use of a large 85 ft. diameter ground antenna for synchronous orbit missions.

Only four different equipment configurations resulted, with the differences caused by whether the vehicle was manned or unmanned or whether long free drift storage in earth orbit was required.

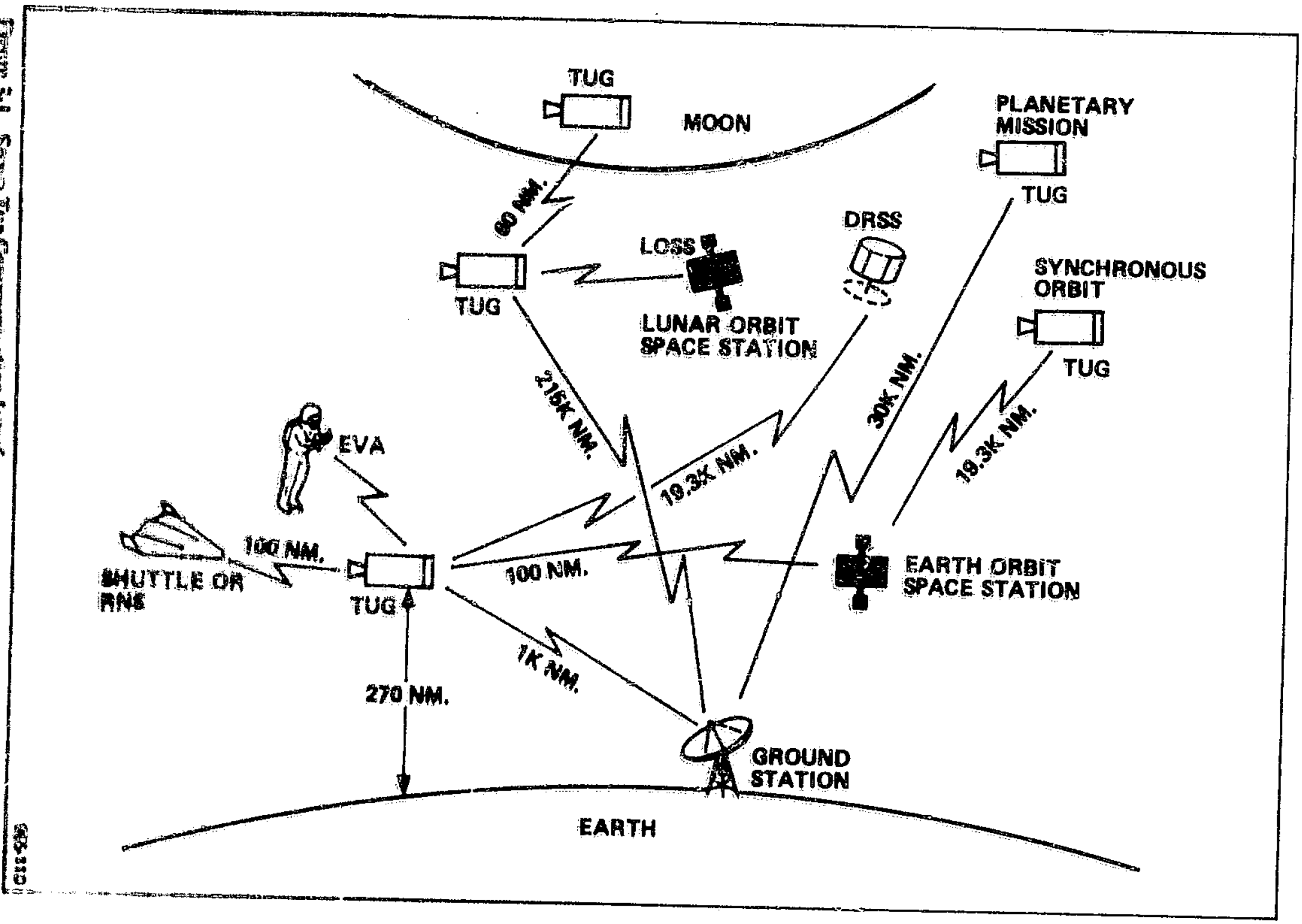

Figure 3-1 Space Tug Communication Interfaces

Table 3-1. Space Tug Communications Requirements Summary

RF Link	Mission	Nominal Carrier Frequency	Channel/Function	Channel Bandwidth	Required Signal to Noise Ratio	Range Nominal Maximum
Tug to MSFN or DSN	All	2.2–2.3 GHz	1 – Digital 1 – Voice 1 – Tracking 1 – TV	1 MHz 3 KHz 50 KHz 2.9 MHz	12 db 12 db 12 db 30 db	~1,000 NM 24,000 NM 210,000 NM
MSFN or DSN to Tug	All	2.1–2.2 GHz	1 – Data/Command 1 – Voice 1 – Tracking	30 KHz 3 KHz 50 KHz	12 db 12 db 12 db	Same as above
Tug to Shuttle Station RNS Loss	Earth Orbit Rendezvous	136 MHz	1 – Digital 1 – Voice	50 KHz 3 KHz	12 db 12 db	~100 NM
Shuttle to Tug Station RNS Loss	Earth Orbit Rendezvous	136 MHz	1 – Digital 1 – Voice	50 KHz 3 KHz	12 db 12 db	Same as above
Tug to DRSS	Earth Orbit	2.2–2.3 GHz	1 – Digital 1 – Voice 1 – Tracking	30 KHz 3 KHz 10 KHz	12 db 12 db 12 db	24,000 NM
DRSS to Tug	Earth Orbit	2.1–2.2 GHz	1 – Digital 1 – Voice 1 – Tracking	30 KHz 3 KHz 10 KHz	12 db 12 db 12 db	Same as above
Tug to EVA	All Manned	136 MHz or ~280 MHz	1 – Voice	3 KHz	12 db	~300 Ft.
EVA to Tug	All Manned	Same as above	1 – Voice 1 – Digital	3 KHz 1 KHz	12 db 12 db	Same as above

A composite communications functional block diagram is illustrated in Figure 3-2. It shows all of the equipment needed for the total tug missions in simplex form.

A Unified S-band system was chosen for the tug-to-ground link because of its availability (space and ground systems), its ability to handle the requirements of simultaneous PCM data, voice, and TV on the downlink, and its ability to provide commands and voice on the uplink. It also provides tracking and ranging data. A high-gain (4 ft. diameter) steerable antenna on the tug appears necessary to maintain the required signal to noise ratio for commercial quality TV on the downlink. A simplex system has tentatively been chosen because of relatively short duration missions (42 day maximum) and the ability to maintain the system by replacement in space.

A VHF voice link is recommended for the manned missions. The redundancy (UHF and VHF) is desirable from a mission point of view and the current planning for the shuttle and space station make use of VHF for the prime vehicle-to-vehicle communications link (Ref. II-6). Thus VHF is required on the tug for compatibility. VHF is attractive for the vehicle-to-vehicle link because it is less susceptible to signal dropouts due to irregularities in the antenna pattern. This will permit reliable voice communications and commands regardless of vehicle attitude. From Figure 4-2, it can be seen that less power is required at a lower frequency (VHF) using omni antennas for a given distance

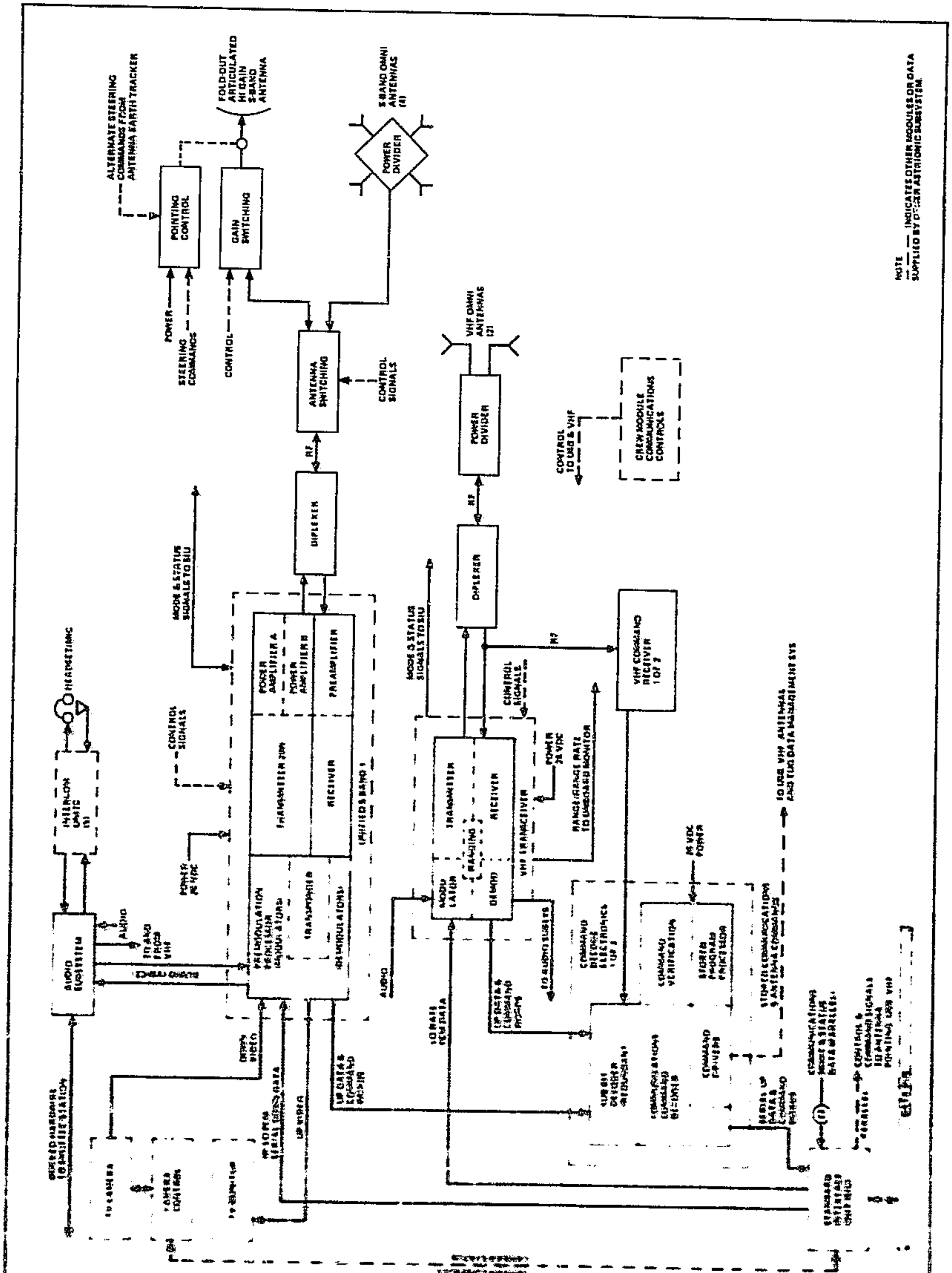

Figure 3-2 Composite Tug Command and Control Subsystem Preliminary Functional Block Diagram

The incorporation of a ranging capability within the VHF system may be required to be compatible with the shuttle and space station.

A Command Decoder Electronic unit is required to perform sub-bit decoding and to provide the interface between communications equipment and the standard interface unit of the data bus. It also performs as the active decoder in the storage mode of operation. Redundancy is planned.

A VHF command receiver is required to receive commands when the tug is in the storage mode. Redundant receivers are planned.

The equipment required for tug communications is listed in Table 3-2. The table gives estimates of the characteristics of size, weight, power, and heat dissipation. The equipment usage on a mission basis is listed along with the total system requirements of weight, power, and heat dissipation.

The communications design should exploit the concept of modular design to make reconfiguration and maintenance in space relatively easy. Most of the communications equipment is small and has a natural interface for removal. Other larger equipment, such as the USB equipment, will be divided into functional units as lowest replaceable units.

The communication subsystem design parameters are discussed in a later section of this appendix. Aside from user requirements, the important design considerations are:

- Bandwidth
- Onboard transmitter power
- Frequency
- Antenna gain (size) at transmitter and receiver
- Factors of power, size, weight
- Cost
- Reliability and maintainability

This study has brought out the fact that better definition of the user requirements for communications is required in order to permit a more detailed design analysis. A number of assumptions were necessary to arrive at the preliminary design proposed in this report.

Table 3-2. Composite Tug Command and Control Subsystem Preliminary Functional Block Diagram

Component Name	Characteristics							Equipment Required							
								Missions							
	Quantity	Size (in.) (each)			Weight (each)	Power (watts)		Sync. Orb. Reuse. Unmanned	Sync. Orb. Expend. Unmanned	Lunar Landing Manned	Low Earth Orbit Manned	Lunar Orbit Manned	Planetary Unmanned	Reusable Nuclear Shuttle Unmanned	Four Stage Saturn V Unmanned
		W	H	D	(lbs)	Act.	Stby.								
Unified S-Band Equipment	1	21	8	7	42	107		X	X	X	X	X	X	X	X
USB Diplexer	1	4	3	1	0,5	–		X	X	X	X	X	X	X	X
USB Antenna Switch	1	2	2	1	0,3	–		X	X	X	X	X	X	X	X
USB Power Divider	1	3	2	1	0,5	–		X	X	X	X	X	X	X	X
USB Omni Antennas	4	6	3	3	0,5	–		X	X	X	X	X	X	X	X
USB Hi Gain Antenna Control	1	10	8	5	8	10		X	X	X	X	X	X	X	X
USB Hi Gain Antenna	1	48		12	12	–		X	X	X	X	X	X	X	X
VHF Transceiver Equipment	1	15	7	5	15	25		–	–	X	X	X	–	–	–
VHF Diplexer	1	4	3	4	1	–		X	–	X	X	X	–	–	–
VHF Power Divider	1	3	2	1	0.5	–		X	–	X	X	X	–	–	–
VHF Omni Antennas	2	2	11	5	1.2	–		X	–	X	X	X	–	–	–
VHF Command Receiver	2	4	6	2,5	2	1	0.5	X	–	–	X	–	–	–	–
Command Decoder Electronics	2	8	5	4	10	5	3	X	X	X	X	X	X	X	X
TV Camera	1	2	6	10	5	6		X	X	X	X	X	X	X	X
TV Camera Control	1	3	3	2	2	3		X	X	X	X	X	X	X	X
Audio Equipment	1	4	2	1	2	3		–	–	X	X	X	–	–	–

4.0 DETAILED ANALYSIS

4.1 SYSTEM DESIGN ANALYSIS

4.1.1 General

The design of communications equipment for a vehicle such as the space tug involves the consideration of a number of factors. Of these, user requirements are foremost. However, only general requirements existed: a command uplink, a downlink consisting of digital data, voice, and TV, provision for tracking and ranging, and vehicle-to-vehicle voice communications. To fill the gap in specific requirements, estimates of bandwidth requirements ranging from 50 KHz for VHF to 10 MHz for Unified S-band were made to use in calculating the communication link parameters.

4.1.2 Design Considerations

The communications design is an iterative process with many implicit trades. Some of these normal system trades are an interaction of:

- Onboard transmitter power
- Bandwidth; determined by data/command bit rates, TV requirements, and number of simultaneous channels
- Frequency
- Onboard antenna size (gain)
- Receiving antenna size (gain)
- Factors of power, weight, and cost are also involved in configuring a satisfactory system

The elements for space tug communications design are summarized in chart form in Figure 4-1. The chart also includes typical ranges of values involved in the communications link calculations. The direction in which some dependent factors will vary with the independent factors are illustrated. For example as the frequency is increased (↑), the path losses will increase (↑), and the antenna size for a given gain will decrease (↓).

4.1.3 Operational Considerations

An understanding of all the operational considerations for the tug is important since these will greatly impact the communication requirements and thus the design. At some later time in the communications design cycle, the initial operational requirements should be examined to see whether too great a penalty is being paid in communications complexity, cost, weight, and power to meet a given operational requirement.

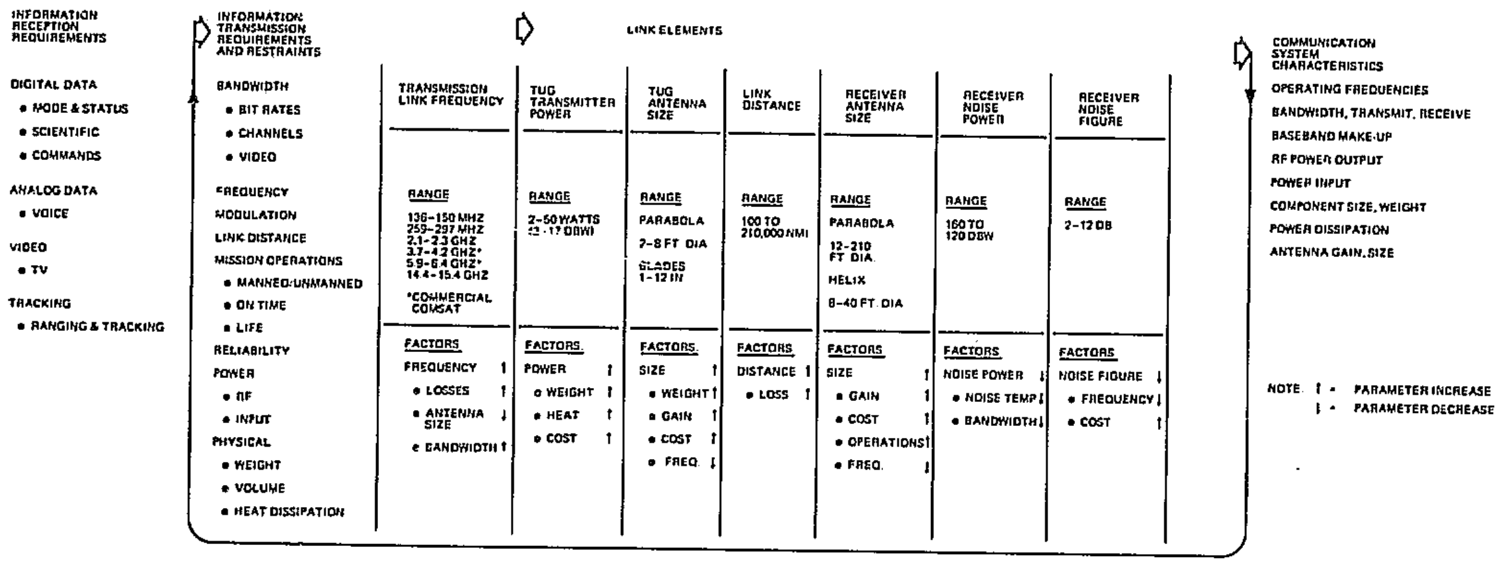

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Figure 4-1. Elements of Space Tug Communications Design

A summary of the tug communications usage is given in Table 4-1 for the synchronous orbit mission. Table 3-1 summarized the tug communications requirements for the RF links involved.

The factor of whether the tug is manned or unmanned for a given mission greatly impacts communications. For the manned versions VHF voice communications (transceiver and intercom) must be provided. Also, some of the functions which could be handled by manual control (antenna control, baseband make-up selection, switching, etc.) in the manned version must be handled by stored onboard programs or remote commands.

Another mission factor which impacts design is whether the tug must be placed in a free drift earth orbit storage mode for 180 days. For this case redundant command receivers and command decoders must be added to process commands to power up the vehicle and bring it out of the storage mode.

4.1.4 Detailed Link Analysis

4.1.4.1 General

Using the concepts outlined by Figure 4-1, the general range equation parameters were calculated or estimated for the range of variables encountered in the tug missions. This was done to establish transmitter power, transmitter antenna gain (size), and receiver antenna gain for a given link distance, frequency, bandwidth, and signal to noise (S/N) ratio.

4.1.4.2 Calculations

The following equations were used in calculating the characteristics of the tug communications system.

Power Budget

$$P_t = P_n + A_p + NF + L_T + L_R - G_T - G_R - NIF + S/N$$

where:
- P_t = transmitter power output in dbw
- P_n = receiver noise power (sensitivity in dbw)
  - P_n = kTB, k = Boltzmann's constant = 1.38×10^{-23} watt-sec.
  - T = antenna noise temperature in ^{o}K
  - B = receiver bandwidth in Hz
- A_p = transmission path loss in db
  - $A_p = 37.8 + 20 \log f + 20 \log d$
  - f = frequency in MHz
  - d = distance in nautical miles
- N_F = receiver noise figure (db)
- L_T = transmit losses (db)
- L_R = receiver line losses plus polarization losses
- G_T = transmitter antenna gain (db)
- G_R = receiver antenna gain (db)
- NIF = noise improvement factor due to modulation techniques (db)
- S/N = required signal-to-noise ratio (db)

Table 4-1. Tug Communication Usage – Unmanned Synchronous Orbit Mission

Mission Phase	Communication Requirements	Communication Channels Used	Channel Description
Prelaunch Checkout – (external to Shuttle)	• Load computer memory • Checkout command, ranging, and data channels • Provide link with GSE for checkout of subsystems. (links with GSE transmitters and receivers may be via hardwire thru umbilical or antenna hats)	All equipment exercised. 1 – Unified S-Band S-Band Xmit. 2.2-2.3 GHz, S-Band receiver 2.1-2.2 GHz 1-VHF 136-138 MHz Command Receiver	USB – 1.5, 5, or 10 MHz bandwidth. Provision for ranging, PCM data, video on downlink. Command/data up. VHF – 50 KHz bandwidth; 200 Hz to 20 kbps. Provision for command up.
Launch–Onboard Shuttle	TBD Depends on whether Shuttle will interface electrically with Tug communication system placed in standby mode.	TBD	—
Earth Orbit Checkout	• Command subsystem ON • Verify command links and data links • Transmit onboard checkout data	All equipment exercised. Hi-gain antennas extended. USB for direct ground comm., USB comm. via DRSS (bent pipe concept possible using hi-gain antenna)	Same as above
Trans-Synchronous	• Commands up, data down • Tracking	USB prime. Ground station tracks USB signal.	Same as above
Synchronous Orbit	• Commands up, data down • Tracking	USB prime. S-Band if video or hi data rates rqd.	Same as prelaunch
Trans-earth	Same as trans-synchronous	—	—
Earth Orbit Rendezvous and Docking with Shuttle or Station	Command up, data down	VHF from ground and/or S-Band to/from ground	Same as prelaunch
Earth Orbit Storage	Command the subsystems to quiescent mode. Tracking requirements for free flying Tug TBD. If Tug docked to Station, communications can be completely powered down.	S-Band powered down. VHF receiver and command subsystem in keep-alive mode	Same as prelaunch

Parabolic Antenna Gain

$$G = 20 \log f + 20 \log D - 52.6$$

where: f = frequency in MHz
D = diameter in feet
G = gain in db

4.1.4.3 Tabulation of Parameters

The following tabulations of paramc...s are useful in understanding the effect or contribution of each parameter in tug communication link considerations. It should be understood that the values used in this study are theoretical or estimates in some cases but should serve as a good starting point for future tug studies.

Figure 4-2 illustrates in generalized terms the effect antenna type, size, and frequency have in reducing transmitter power required.

Table 4-3 lists typical tug antenna configurations and the net gain resulting from the combinations with various ground receiving antennas.

Table 4-4 lists the free space path loss for the several anticipated tug communication links.

Table 4-5 lists typical receiver noise figures for the receivers involved in the tug communication links.

Table 4-6 lists theoretical receiver noise power for various bandwidths. A noise temperature of 300°K was used but the temperature can vary from 30°K to 3000° K depending on the direction the antenna is pointing. The temperature used for antennas looking at the earth is usually 290°K. Together the receiver noise power (sensitivity) and the receiver noise figure determine the receiver input power required.

4.1.4.4 Results

A unified S-band communications link was chosen as the prime link for tug communications. It can provide the required bandwidth and signal-to-noise ratios and is already is use in many space programs. Its current use means usable equipment already exists, and second generation improvements are realizable within the tug development cycle time.

The advantage of the UHF Unified S-band frequencies (2.1-2.3 GHz) over VHF is the increased gain attainable with reasonable-size antennas (2 to 4 feet diameter) and the existence of ground facilities operating at S-band. The advantage of S-band over SHF Ku-band (13-15 GHz) is the considerably lower atmospheric attenuation at S-band. The requirements for and the implementation of a Ku-band link for tug to DRSS needs further investigation.

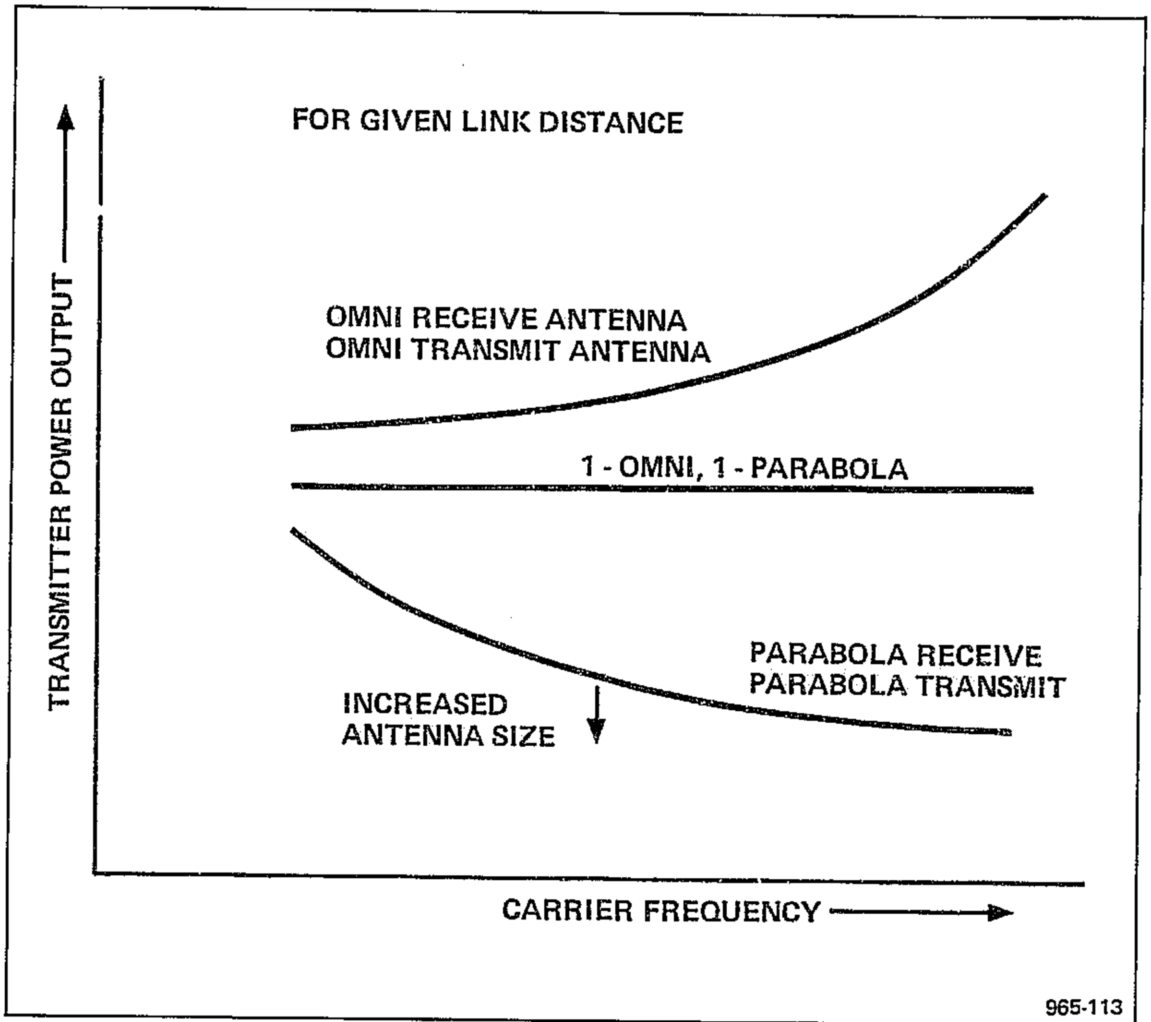

Figure 4-2. Generalized Antenna Performance Curves

Table 4-3. Typical Tug Antenna Configurations and Net Antenna Gains

Nominal Link Frequency	Tug Antenna			Communicating Station Antenna				Net Antenna Gain or Loss (db)
	Type	Size	Gain (db)	Name	Type	Size	Gain (db)	
VHF (136 MHz)	Omni	1 ft.	-3	Shuttle or Station	Omni	1 ft.	-3	(-6)
	Omni	1 ft.	-3	MSFN	Multi-helix	8 ft. dia.	+15	+12
	Omni	1 ft.	-3	DRSS	Yagi	5 ft.	+16	+13
UHF (2.2 MHz)	Omni		-3	MSFN	Parabola	30 ft. dia.	+43	+40
	Omni		-3	DSN	Parabola	85 ft.	+53	+50
	Omni		-3	DSN	Parabola	210 ft.	+61	+58
	Omni		-3	DRSS	Parabola	8 ft.	+32.3	+29.3
	Parabola	2 ft. dia.	+20.3	MSFN/DSN	Parabola	39'/85'	43'/53'	+63.3/+73.3
	Parabola	4 ft. dia.	+26.3	MSFN/DSN	Parabola	30'/85'	43'/53'	+69.3/+79.3
	Parabola	8 ft. dia.	+32.3	MSFN/DSN	Parabola	30'/85'	43'/53'	+75.3/+85.3

Table 4-4. Path Loss for Tug Communication Links

Nominal Link Frequency	Link Distance (NM–Max.)	Path Loss (db)	Mission Use (Tug Location and Contact)
VHF (136 MHz)	100	120.5	Separation and Rendezvous Communications with Shuttle, Station, Tug, or EVA
	1,000	140.5	LEO Communications with MSFN or AF SGLS
	24,000	168.1	LEO Communications with DRSS or TACSAT
UHF (2.2 GHz–Unified S-Band)	100	144.6	Vehicle to Vehicle; Tug to Tug, Shuttle, or Station
	1,000	164.6	LEO Communications with MSFN or AF SGLS
	24,000	191.2	LEO Communications with DRSS or TACSAT
			Sync Orbit Communications with MSFN or AF SGLS
	30,000	194.1	Planetary Mission Communications with MSFN
	210,000	211.0	Lunar Missions with DSN (Earth)
SHF (14 GHz)	24,000	210.2	LEO Communications with DRSS

Table 4-5. Typical Receiver Noise Figures

Link	Receiver	Noise Figure
VHF (136 MHz)	Tug/Shuttle, Station	10-12 db
	DRSS/TACSAT	10-12 db
	MSFN/AF SGLS	8-10 db
UHF (2.2 GHz)	Tug/Shuttle, Station	4-6 db
	DRSS/TACSAT	4-6 db
	MSFN/AF SGLS	2-4 db
SHF (14. GHz)	Tug	6-8 db
	TDRS	4-6 db

Table 4-6. Receiver Noise Power

Bandwidth	Receiver Noise Power (kTB); T = 300°K
50 KHz	-156.8 dbw
1.5 MHz	-142.1 dbw
5 MHz	-136.8 dbw
10 MHz	-133.8 dbw

The example power budget of Table 4-7 shows that a 20 watt onboard tug transmitter in conjunction with a 4 ft. diameter antenna on the tug (pointing at the ground station) and an 85 ft. diameter ground receiving antenna (tracking the tug) is sufficient for the transmission of commercial grade TV.

Twenty watts of transmitted power is easily realizable and equipment size and weight are reasonable. It should be noted that only a 3 db gain is obtained by doubling the output power, and this method of increasing performance is costly in terms of dollar cost, onboard weight, and power dissipation. The manipulation of antenna size and frequency is a more practical way to obtain required signal-to-noise ratios.

It may be noted in the example of Table 4-7 that polarization losses and noise improvement factors (NIF) were not included. These would not likely affect the overall results of this brief preliminary study.

The power budget for the VHF tug-to-shuttle/space station 100 nmi. link was calculated (see Table 4-8). It shows that a 2 watt transmitter on the tug would be adequate for this application. Omni antennas were assumed for both vehicles. The role that VHF is to play as a back-up for the USB voice and command links needs further definition and analysis.

This study has revealed the requirement for a low power VHF command receiver for reception of commands during the storage phase. This receiver could share the VHF antenna system used by the VHF transceiver for the manned missions.

Other link margins or power budgets were calculated in conjunction with this study but are not included here for brevity. It is recognized that this study is not complete, but one objective was merely to illustrate the factors involved in the conceptual design of tug communications.

A study objective was to configure a tug communications layout to permit engineering estimates of size, weight, and power requirements. The results are summarized in Figure 3-2 and Table 3-2.

Table 4-7. Unified S-Band Tug-to-Ground Link Calculations (Sync Orbit Mission)

Parameter	Nominal Value
Tug Transmitter Power (20w)	+ 13.0 dbw
Transmit Line Losses	- 2.5 db
Tug Antenna Gain (4' Dish)	+ 26.3 db
Free Space Path Loss	- 191.2 db
Ground Antenna Gain (85' Dish)	+ 53.0 db
Receiver Line Losses	- 2.5 db
Received Carrier Power	- 103.9 dbw
Receiver Noise Power (kTB)	- 136.8 dbw
Receiver Noise Figure	+ 2.0 db
Required Power to Receiver	- 134.8 dbw
Signal-to-Noise Ratio (S/N)	30.9 db
Required S/N for TV Transmission	30.0 db
Signal Margin	+ 0.9 db

Notes:
Distance: 24,000 NM (Sync Orbit Max Distance)
Frequency: 2.2 GHz (Nominal USB)
Bandwidth: 5 MHz (2.9 MHz TV plus Down Data)

Table 4-8. VHF Tug-to-Shuttle* Link Calculations

Parameter	Nominal Value
Tug Transmitter Power (2w)	3.0 dbw
Transmit Line Losses	- 2.5 db
Tug Antenna Gain (Omni)	- 3.0 db
Free Space Path Loss	- 120.5 db
Shuttle Antenna Gain	- 3.0 db
Receiver Line Losses	- 2.5 db
Received Carrier Power	- 128.5 dbw
Receiver Noise Power (kTB)	- 156.8 dbw
Receiver Noise Figure	12.0 db
	- 144.8 dbw
Signal-to-Noise Ratio (S/N)	16.3 db
Required S/N	12.0 db
Signal Margin	+ 4.3 db

Notes:
*Shuttle, Station, or Tug
Distance: 100 NM
Frequency: 136 MHz
Bandwidth: 50 KHz

4.2 SYSTEMS DESCRIPTION

4.2.1 General

The communication elements for tug have been configured as a result of this study to meet the anticipated requirements for several tug missions. This is a conceptual design based for the most part on existing equipment capabilities. In some instances, existing equipment (such as Apollo CSM and LM) characteristics were used as guidelines for estimating the size, weight, and power of the equipment outlined here. As mentioned earlier, this preliminary system configuration was necessary in order that the overall tug astrionic module design could take place and be meaningful.

4.2.2 System Configuration

A composite communications functional block diagram is illustrated in Figure 3-2. This block diagram shows the total system equipment needed for communications. The equipment as shown is actually needed for only the low earth orbit (LEO) manned mission which is followed by the 180 day free drift storage. The other missions require a lesser amount of equipment as shown by Table 3-2.

The functional block diagram illustrates the flow of the information signals and the control signals. The interface between the communications equipment and the other subsystems for digital signals is via the standard interface unit (SIU) and data bus. Power for communications is 28 vdc supplied by the astrionic module.

The equipment shown as dotted boxes represents equipment which would not normally be located in the astrionic module. It is also conceivable that the equipment used only for manned missions could be located in the crew module.

4.2.3 Equipment Description

4.2.3.1 General

A listing of tug communications equipment characteristics (size, weight, power, heat dissipation) together with equipment usage is given in Table 3-2. The totals are given for each mission for each characteristic itemized.

It should be noted that only four different equipment configurations resulted from this study. These differences resulted from the different requirements between manned and unmanned vehicles and from the requirement for free drift orbital storage at the end of some missions. The manned versions require extra equipment in the form of a VHF transceiver for backup voice communications and an audio subsystem. The storage mode missions require additional redundant command receivers and command decoders.

It is conceivable that a tug may be reconfigured for a new mission while docked to a shuttle or space station. The "modular" design concept for tug astrionics would preclude the necessity of flying excessive capability on any one mission.

4.2.3.2 Unified S-Band Equipment

The Unified S-Band (USB) equipment is the heart of the communications layout since it is the prime link between the tug and the ground stations.

The USB link was chosen because of the anticipated requirement for simultaneous downlink TV, PCM, and voice (for manned missions). The USB also provides the uplink command channel and transponder for tracking and ranging.

The equipment consists of a transmitter, a power amplifier (tentatively identified at 20 W), a receiver with preamplifier, a transponder for turn-around of the ranging signal, a premodulation processor providing subcarriers and baseband make-up, and the demodulators for the uplink signals. The power amplifier could be modular with high and low power outputs selectable as required.

The input signals, in addition to controls and power, are: RF input to receiver, PCM down data, video, and audio to the premodulation processor.

The output signals, in addition to the mode and status signals, are: RF to the antenna system, updata (PCM) to the command decoder, and audio to the audio subsystem.

The estimated characteristics of the USB equipment are listed in Table 3-2.

4.2.3.3 Unified S-Band Antenna System

The S-Band RF signal flow (out) is from the transmitter through the diplexer to an antenna switch which permits selection of the omni coverage lo-gain antennas or the directional hi-gain antenna. The omni antennas (4) are conventional flush mounted types connected through a power divider.

The hi-gain antenna has tentatively been identified as a four-foot diameter parabola providing a gain of approximately 26 db. It may be necessary to have a gain switching feature (beam width changing) for this antenna. The antenna is to be articulated with provision for using pointing signals developed by the tug data processor to point the antenna toward the station communicating with the tug. This steerable type antenna is necessary in order not to place attitude constraints on the tug while communicating on links requiring the use of the hi-gain antenna.

4.2.3.4 VHF Transceiver Equipment

A VHF transceiver will be required as a back-up for voice and low data rate transmissions for manned missions. The VHF link will be the prime voice link between the tug and the shuttle, station, or other vehicles. Its omni antennas relieve the tug of any attitude restrictions while docking, maneuvering, and communicating with space vehicles. The omni antennas (2) are a blade type and will extend a few inches out from the skin surface. They are coupled to the RF cable and diplexer through a power divider. Calculations show that a 2 watt transmitter would be sufficient for 100 nmi. vehicle-to-vehicle communications.

In addition to the voice link, the VHF transceiver could provide a low data rate channel with a bandwidth of approximately 30 to 50 KHz. Possible carriers are in the 136-148 MHz range.

For some missions there may be a requirement for a VHF range and range rate transponder to be compatible with shuttle and space station. This feature can be provided by the VHF equipment.

The EVA communications are also in the VHF band with possible frequency assignments in the 259 to 297 MHz range.

4.2.3.5 Audio Subsystem

The audio subsystem is required for manned missions and provides the line drivers and interface conditioning between the intercoms (head sets) and the modulators/demodulators of the transmitter/receiver.

For the docked mode, a hardwire interconnection between the tug and the other vehicle has been anticipated.

The number of intercom units has been estimated at five: 3-crew module, 1 - astrionic module, and 1 - cargo module.

4.2.3.6 VHF Command Receiver

The VHF command receiver is required as a low power drain device to receive commands to power-up the astrionic module and bring other subsystems on. This requirement results from the 180 day free drift storage mode.

The receiver will operate in the 136-148 MHz band and is a relatively small, light weight, and low power package (see characteristics, Table 3-2). It utilizes the VHF omni antennas mentioned in Section 4.2.3.4.

To meet reliability requirements it is anticipated that two of these units will be provided which will operate at separate frequencies.

4.2.3.7 Command Decoder Electronics

The purpose of the command decoder electronics is to perform the function of a sub-bit decoder and to interface with the standard interface unit (SIU) to put the uplink command words on the data bus. Whether the functions of message validation, parity checks, etc. are performed in this unit will depend on the philosophy of the total data bus design, the quantity of commands to be processed, and the data rates.

For certain commands relating exclusively to communications, it is considered that the unit will perform the decoding, verification, and command driver functions. The commands to bring the astrionic module out of the storage mode will also be processed by the command decoder electronics. Reliability requirements will be met by providing redundant circuits or units as required.

4.2.3.8 TV Equipment

The TV equipment is treated as part of communications even though it will probably be located in the crew module, carge module, or both. Commercial grade black and white TV requiring 2.9 MHz bandwidth has been postulated. (Color will require a 4.2 MHz bandwidth.)

The camera and its control unit will be space qualified items capable of remote control through the command link. The control unit will incorporate all the normally desired features of lens and filter changes, zoom, and complete articulation.

The monitor unit is provided for monitoring the onboard camera output. It is treated as part of the display and monitor equipment.

4.2.3.9 Communications Controls

The controls for the communications equipment for the manned missions will be located in the crew module. They are therefore considered as part of the display and control equipment. The exact functions which will require crew control and the exact method of implementing the controls have not yet been established.

4.3 SYSTEM CONSIDERATIONS

4.3.1 Environment

In a study of this nature some consideration must be given to environmental factors. With the exception of the reusable nuclear shuttle (RNS) no new environmental conditions have been identified in which space communications equipment has not already been proven reliable. Meeting the RNS environmental conditions, when they become known, is not expected to be a major problem for the communication equipment.

The communications equipment outlined in this study will operate within the temperature range of -20°C (-4°F) to +75°C (+168°F) and this could be extended if required by additional development and qualification effort. In a storage mode a typical range is -40°C to +85°C.

4.3.2 Reliability

The reliability of all the tug communications equipment is estimated at 0.996 for a 14 day mission. This figure has been associated with the Apollo USB equipment. Considering possible improvements within the tug development time frame, it is expected that this figure could be improved even for longer missions.

A simplex configuration is recommended for the communications equipment with the exception of the command receiver and command decoder primarily required for the 180 day free drift storage mode.

4.3.3 Onboard Checkout

The communications equipment must have sufficient test and monitor points built-in to permit automated checkout and test in conjunction with an onboard checkout function. The function should isolate faults to a replaceable unit for ease of maintenance and quick turn-around repair by replacement.

4.3.4 New Technology Considerations

Techniques and hardware now exist for meeting the anticipated tug communications requirements as evidenced by Apollo CSM and LM, Lunar Orbiter, Mariner vehicle, and synchronous satellite operation. The tug, however, should employ second generation USB equipment to take advantage of expected technology advancements. These improvements will be evolutionary in nature. Small but significant improvements in transmitter efficiency to reduce onboard power requirements, in electronic circuit packaging to reduce size and weight, and in reliability to improve operations can be anticipated in the 1971 to 1974 time frame.

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APPENDIX I

ANALYSIS OF ASTRIONIC MODULE STRUCTURE
AND COMPONENT PACKAGING AND LAYOUT

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

This section deals with packaging schemes for housing, supporting, and protecting the equipment in the astrionic module (AM). Packaging constraints at the "black box" level are presented as well as the mounting of boxes to panels, mounting of panels to the structure, structural options, and interfacing of the astrionic module with other tug elements.

Primary emphasis has been placed on determining the feasibility of locating the necessary complement of astrionic equipment in the physical space allotted. Equipment panel layouts were derived and methods of integration of panels to the various potential structural concepts were investigated. Prime consideration was given to accessibility, weight, and modularity of design.

Potential astrionic module structural options were investigated relative to interface and structural loading requirements. Two potential schemes, the open frame and shell structural concepts, are presented and discussed in the study.

2.0 STUDY GUIDELINES AND GROUNDRULES

The basic concepts developed for the astrionic module structure and equipment packaging and layout were based on the following guidelines and groundrules:

- The astrionic module shall be of modular design to optimize the use of common equipment and concepts across the spectrum of space vehicle element requirements.
- The astrionic module will be constructed in a basic ring configuration which is nominally 14 feet in diameter with height to be optimized.
- The astrionic module will be configured to allow maximum accessibility for remove-and-replace maintenance and reconfiguration operations in space.
- A standard physical interface for astrionic equipment will be provided to simplify interchange of components.
- The space tug shall be compatible with the earth and lunar orbiting space stations, the space shuttle and the reusable nuclear shuttle.
- The space tug shall have docking devices compatible with all space vehicle hardware elements.
- Personnel safety shall be maximized during maintenance and reconfiguration operations.
- The tug may be launched either in a four stage Saturn V configuration or in the cargo bay of the space shuttle. In either case, the tug will be subjected only to acceleration loads from powered flight (i.e., no aerodynamic loading).

3.0 SUMMARY OF RESULTS

- Using present "state-of-the-art" black box level packaging, all of the astrionic equipment, including expendables, required for the most complex tug mission can be mounted within 13.5-feet diameter by 4-feet high astrionic module envelope.
- If reliability enhancement techniques dictate extensive black box level redundancy, component installation design is impacted, and the above conclusion would require further study.
- In most cases, mission peculiar (unique) subsystems or equipment can be located on a single panel allowing total panel modularity of installation and removal.
- Reconfiguration of tug equipment complement via extra-vehicular activity (EVA) in space is feasible.
- Two structural concepts, shell and open frame, are feasible and should be evaluated further as loads become better defined and impact on interfacing structures can be evaluated. Several variations of providing adequate accessibility are available with each concept.
- AM structural weight is relatively insensitive to tug payload weight, indicating that one structural design for all tug missions appears attractive.
- Micrometeoroid protection requirements appear to be the most significant environmental factor in the design of the AM structure.

4.0 DETAILED ANALYSIS

4.1 COMPONENT HOUSING CONCEPT

The majority of the astrionic equipment to be used in the astrionic module will be present "state-of-the-art" in function; however, the application of the equipment to the space tug mission requirements will call for special or modified packaging. This is standard procedure for space oriented hardware and can be assumed to add little cost over other space oriented hardware of similar functional capabilities.

To satisfy the space tug mission requirements, the components (black boxes) will have some unique characteristics. The prime factors used for guidelines for the component housing concept are - crew safety, EVA time, equipment function, and cost. The concept recommended is one of packaging the required equipment in the optimum number of housings possible within a maximum size which is constrained by physical space, handling, and maintenance requirements. This concept of packaging entire functional elements, or subsystems, in as few housings as possible enhances the ability to customize the astrionic module for each particular mission. In addition, interconnections and system weight are reduced to a minimum.

4.1.1 Shape

The shape of the black boxes will be such that they will have sufficiently rounded corners and edges to prevent puncture or tear to personnel pressure suits. Boxes will have a hinged cover, as required, capable of latching/unlatching by a gloved hand. Hinges and

latches will be capable of maintaining a pressure sealed condition when the covers are closed. These hinged covers will allow access to the lowest replaceable units (LRU) which can be at a level internal to the boxes.

Hardware used for mounting the units on the panels/structure will be operable by the gloved hand with minimum tooling. The panel contact area (back) of the boxes is flat and of such an area that maximum heat transfer exists consistent with physical layout constraints.

4.1.2 Size

All black boxes will be of a formulated width and height and will have mounting hardware located to match two or more holes in the component mounting panel. This will allow installation of any box in any area containing sufficient holes for mounting without interference with adjacent equipment (see Figure 4-1). The size of any component to be handled shall not exceed 20 x 25 x 40 inches and shall have a mass moment of inertia less than or equal to 250 lb-in-sec^2.

4.1.3 Handholds and Tether Attach Points

Each black box shall have at least one handle, compatible with the gloved hand, located on the face of the cover so as to enable the correct positioning of the unit on the mounting panel. In addition to the handle(s), the boxes shall have a tether attach ring or eye recessed in the cover.

4.1.4 Electrical Interface

Each component unit will have all electrical connectors on the cover recessed below the mean surface plane. These connectors will provide interface to other components, the data bus, and the power system, as required. The electrical connectors shall be a quick-disconnect type capable of operation by the gloved hand without the use of tools. Connectors will be keyed to avoid potential cross-connection during removal and/or replacement operations.

4.1.5 Mechanical Interface

All mechanical pressure and fluid connections will be located on the component cover and recessed below the mean surface plane. The connectors will be a quick-disconnect type capable of operation with the gloved hand without the use of tools. Pressurized components will be fitted with automatic pressure release devices actuated by disconnecting the pressure line.

4.1.6 Component Marking

All component units shall be clearly marked for:

- Component identification with part number, part name and weight
- Location in astrionic module (panel no./panel location)

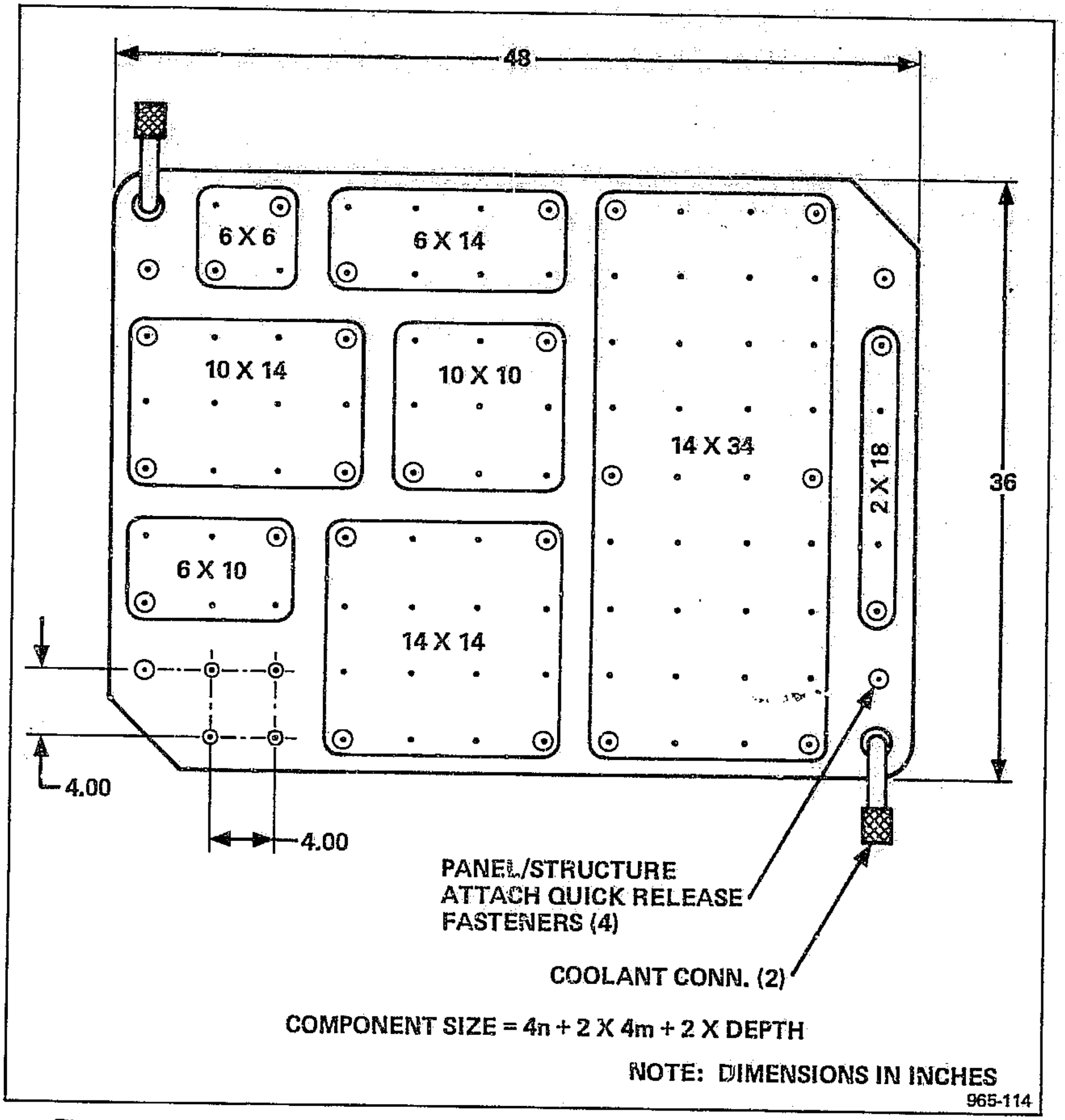


Figure 4-1. Component Mounting Panel (Conditioned)

- Cable connection identification
- Pressure requirements
- Warning labels, if required
- Orientation

4.2 COMPONENT MOUNTING PANELS

4.2.1 Panel Description

To fulfill the requirement for maximum modularity, the component mounting panels shall be one size and shall be interchangeable. A preliminary size of approximately 36 by 48 inches has been chosen. The size of the panels may be reduced with the reduction in size of the final component design with coincident reduction in module weight.

The panels will be rectangular in shape and will have all corners, edges and protrusions rounded sufficiently to allow handling in space without puncturing or tearing personnel pressure suits.

Two types of panels will be used: conditioned panels for components requiring thermal control and unconditioned panels for equipment containing an integral thermal control or requiring no thermal control. Both types will be physically interchangeable, having the same mounting holes for installation to the structure and the same hole patterns for attaching components.

For the components requiring thermal conditioning, panels with fluid passages will be used. These panels will have a quick-disconnect fitting for a plug-in type connection with the thermal control manifold. The coolant fluid will be contained in the panels at all times whether in use or in storage to minimize the entry of air or voids into the system when changeout occurs. This concept also reduces the requirement to refill the coolant reservoirs.

Components having no need for thermal control or having internal conditioning will be mounted on unconditioned panels. These panels will be lighter in weight than the panels with coolant passages. The prime advantage offered by the unconditioned panels is cost saving over the conditioned panels.

Each panel will be fitted with captive quick-release type fasteners used to mount the panels to the structure. The fasteners shall be operable by the gloved hand with minimum tooling.

All panels shall have identical hole patterns for the mounting of components (see Figure 4-1). A hole pattern of 4 by 4 inches over the panel surfaces is a feasible pattern for attaching the components by means of latch type fasteners. This will allow a random placement/non-interference installation of components. The optimum hole pattern may be determined by heat transfer capabilities of contact areas between components and panels.

Each panel will have the capability of accepting a detachable handhold to be used during changeout operations. The handhold shall attach and detach with the gloved hand and with minimum tooling.

All panels shall be clearly marked for:

- Identification by part number, name and weight
- Location in astrionic module (Panel No.)

- Coolant connection inlet and outlet
- Warning labels, if required
- Orientation (top)

In keeping with the modular concept, it is usually feasible to group related components on a panel to minimize the changeout time and effort for reconfiguration. For example, some missions will not require complete navigation systems; others may not require complete power systems. With the proposed packaging scheme, these may be added or deleted with the changeout of a complete panel.

Figures 4-2 and 4-3 depict the astrionic equipment layout as defined for the lunar landing mission. All other missions would result in a deletion of equipment from this layout. As can be seen, entire subsystems can be grouped on individual panels. All necessary equipment can be housed in the required space with an 8-panel configuration.

4.2.2 Mounting Pads, Brackets and Hinges

The component mounting panels will be attached to the structure by means of pads, brackets or hinges permanently fixed on the skin or structural frames. In addition to the pads, brackets or hinges required to mount the component panels, other pads and brackets shall be fixed to the structure to accommodate the mounting of components directly to the skin or structural frames between the panels or on the external surfaces. All pads, brackets and hinges shall accept the same captive, quick-disconnect fasteners used on the panels for attaching components.

4.3 STRUCTURE

4.3.1 Design Requirements

The astrionic module structure is defined totally in terms of the satisfaction of system requirements. These requirements define the functions which the structure must perform and under what conditions this performance is to be accomplished. Functions and conditions include:

- Loads Environment
- Equipment Mounting
- Accessibility
- Environmental Protection
- Weight
- System Growth
- Modular Interface

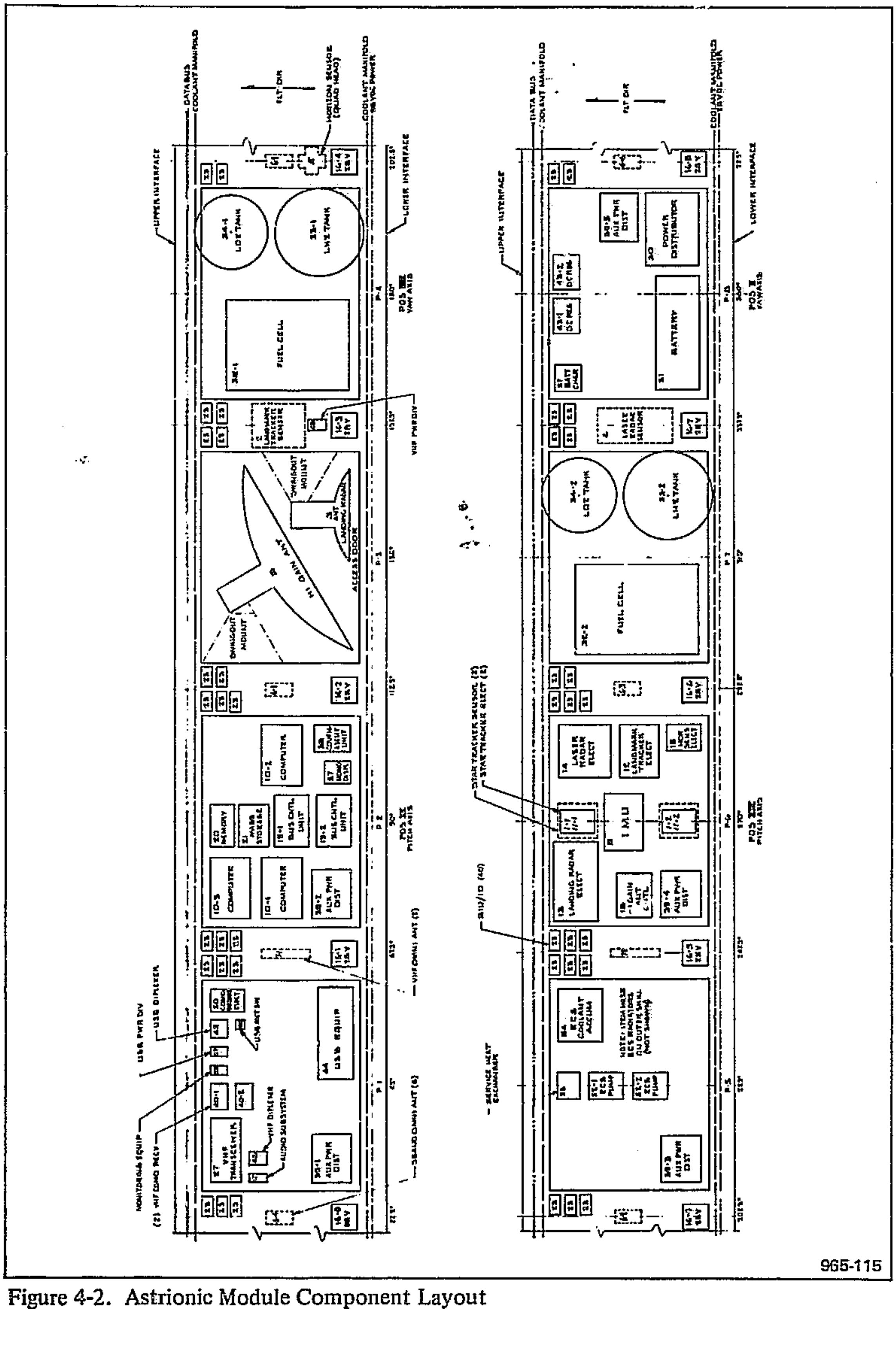


Figure 4-2. Astrionic Module Component Layout

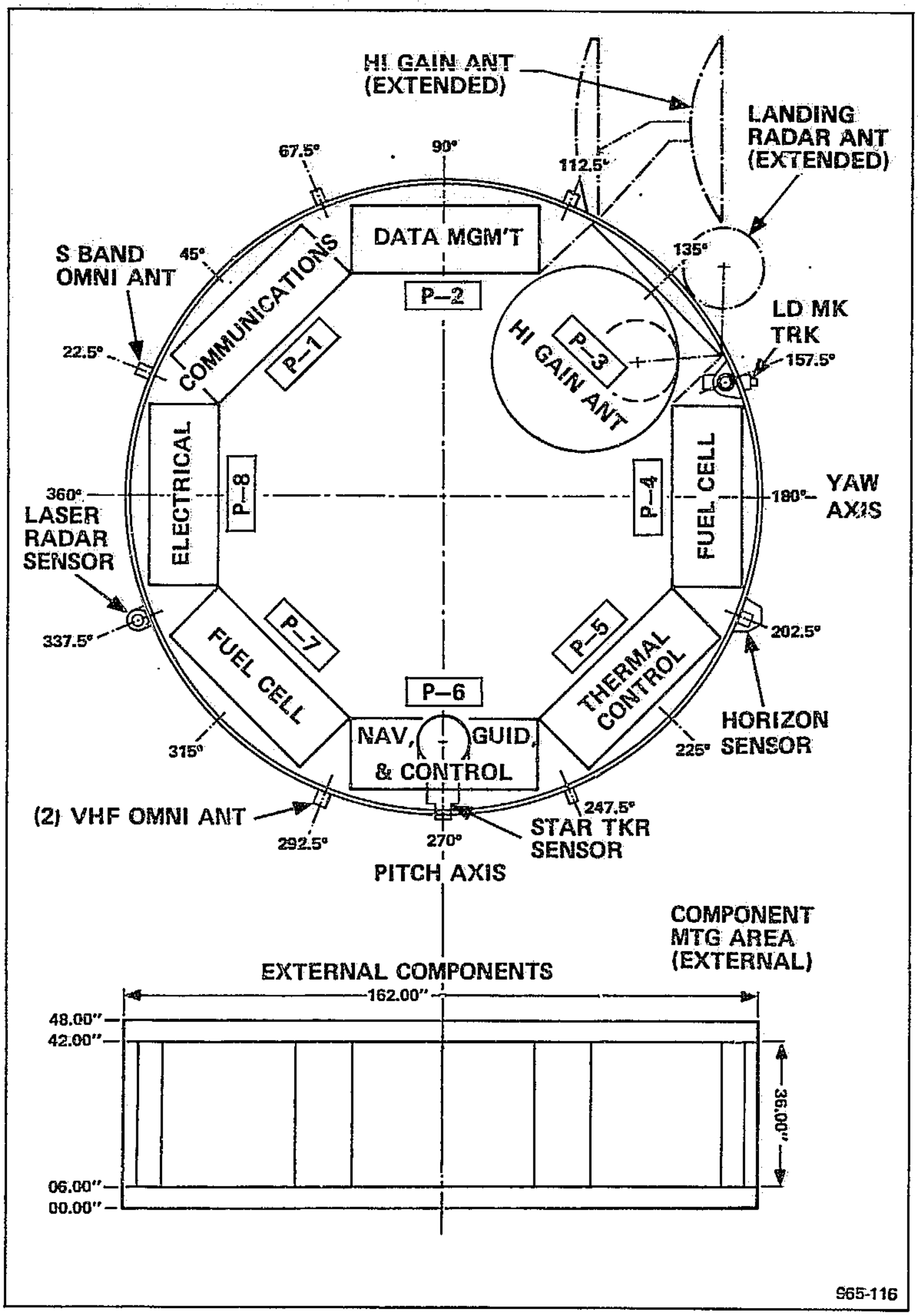


Figure 4-3. Astrionic Module Component Layout (Top View)

4.3.1.1 Loads Environment

Several ground rules for determining preliminary loads for the astrionic module were formulated.

- For four stage Saturn V applications, the astrionic module will not transmit boost flight loads to any "payload" except for space tug elements above the module. To ensure investigating the worst loading case, it is considered possible that the space tug will be launched either nose-up or nose-down.

- A series of launch stack loads ranging from 40,000 to 120,000 pounds was used to develop curves for structural weight.

- The dynamic load factors are similar to those of the Saturn V. The peak load factor considered is 5.9 g; Saturn V steady-state load factors are in the 4.0 to 4.5 g range. In this preliminary structural investigation, no effort has been directed to a better definition of the peak load factor.

- The ultimate load factor used for this study was set at 8, derived from 1.4 manned rating and 5.9 peak dynamic loads. The actual factor is 8.26 but has been rounded off for the study.

- It is not necessary for the loads into and out of the astrionic module to be uniformly distributed. The module has a limited depth, and in most cases the adjacent structure has more depth and therefore a more efficient means of redistributing concentrated loads.

- The four configurations of launch attitude and support methods as a four stage Saturn V configuration, shown in Figure 4-4, represent "worst-case" support conditions for the tug. Neglecting the weight of the tug and using propulsion module and payload weights as shown in the figure, a "stack load" of 100,000 pounds is defined.

4.3.1.2 Equipment Mounting

The equipment loads on the astrionic module are planned to be transferred to fittings provided on the structure. These fittings are attach points for the eight equipment panels. The magnitude of the loads introduced is usually small enough that only local effects need be considered. Dynamic effects will require a more detailed study and can best be accomplished using an actual configuration with actual equipments.

4.3.1.3 Accessibility

Accessibility requirements frequently have significant impact on structural design. For example, the provision for a door in a structure requires considerable reinforcement around that door to be structurally adequate for the defined load environment. Furthermore, the creation of stress concentrations causes problems in structural fatigue which must be addressed. A diagram of the astrionic module structure with large cutouts as compared to a "baseline" honeycomb structure is shown in Figure 4-5.

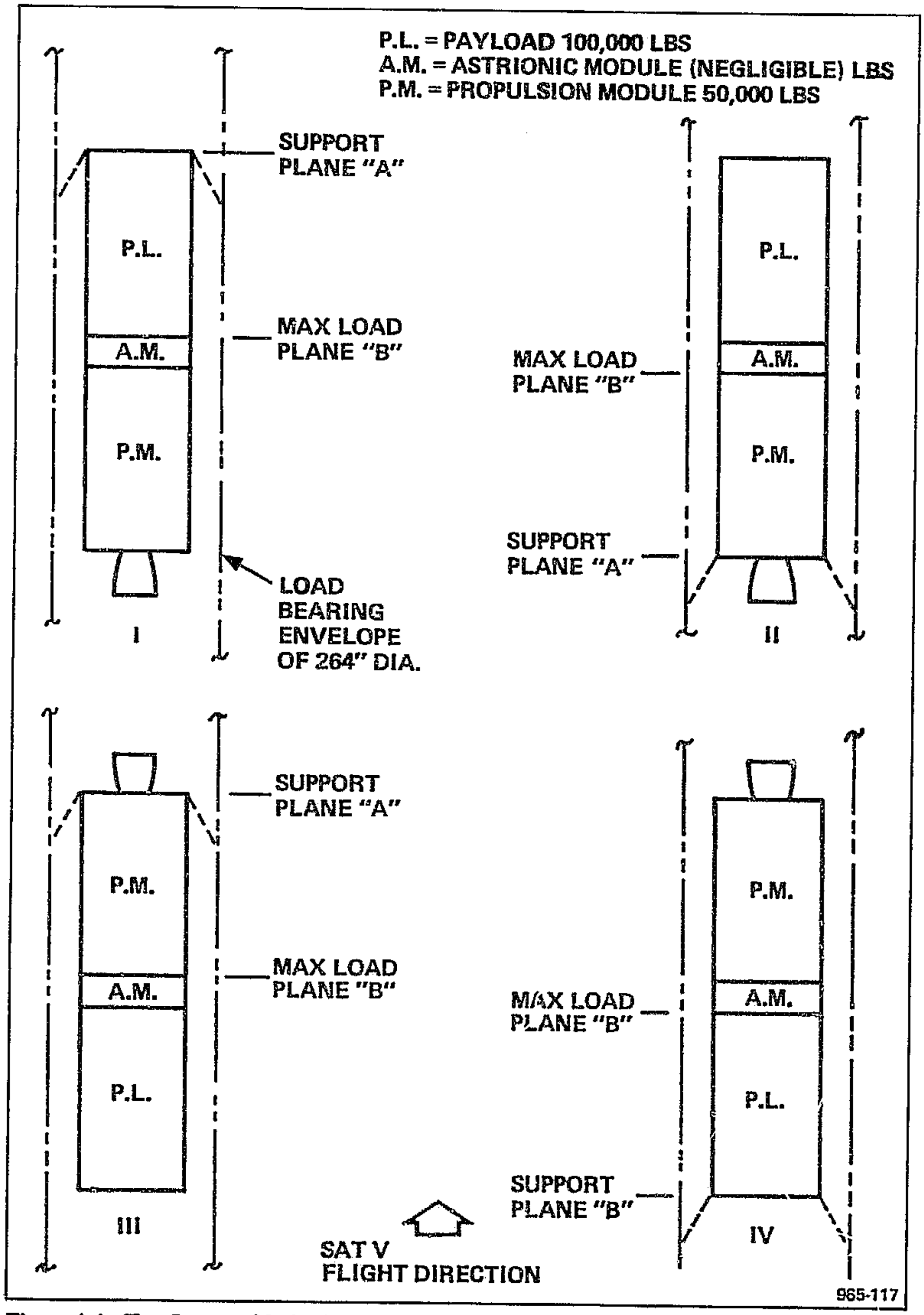


Figure 4-4. Tug Support Methods

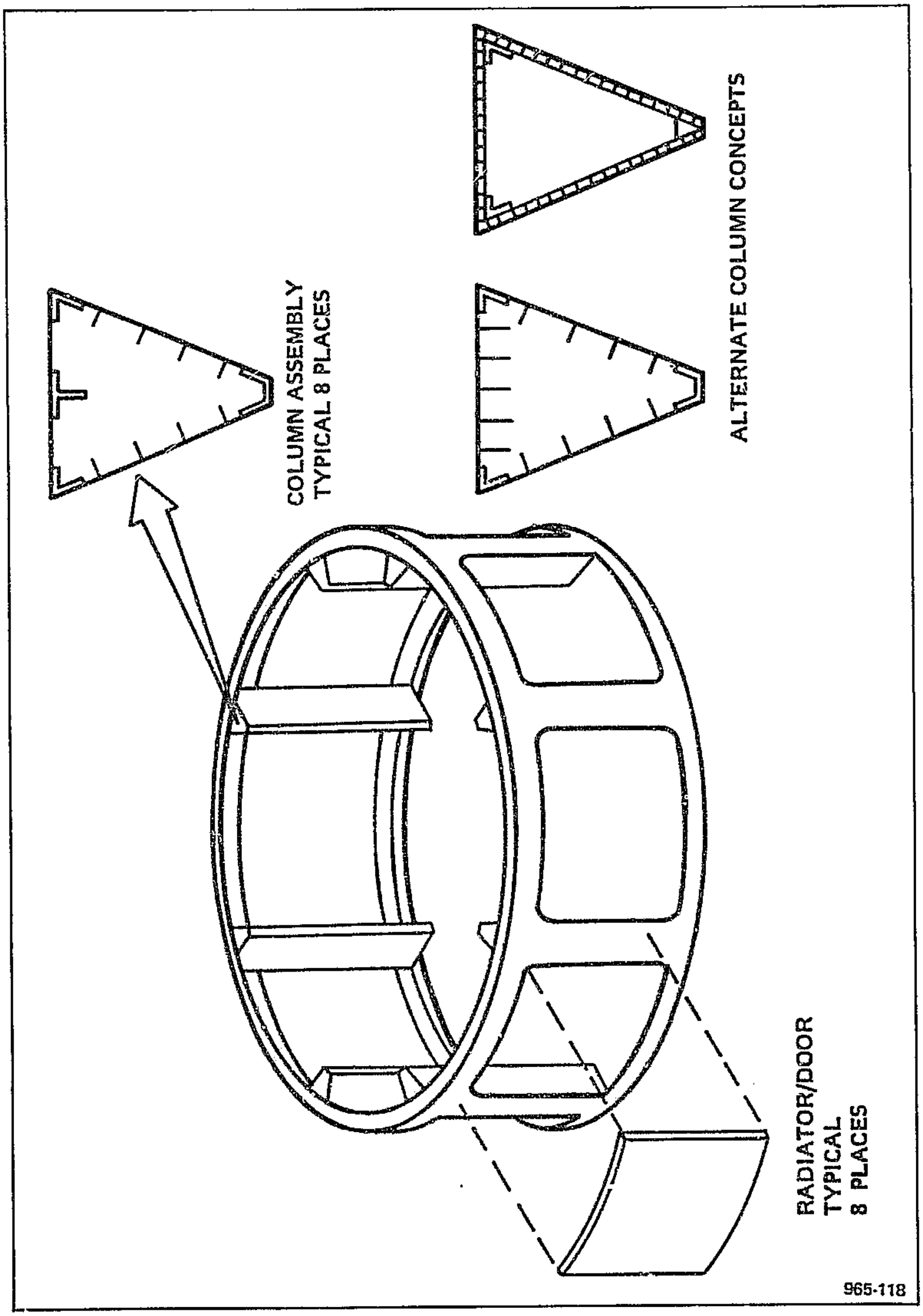


Figure 4-5. Structural Concept, Open Framework

4.3.1.4 Environmental Protection

Environmental factors considered in the definition of the structure include: (1) natural radiation, (2) boost flight acceleration loads and (3) micrometeoroid protection. Since the astrionic module will not carry primary flight boost vehicle loads (reference I-1), but will be carried inside a shroud during boost, the effect of item (2) above is minimized.

Metallic materials are not adversely affected by natural radiation expected to be encountered in space operations (references I-2, I-3, I-4). The leading candidates for structural materials for the astrionic module are metallic and are considered adequate for this environment. A more thorough discussion of radiation impact on the astrionic module is found in Appendix P.

Micrometeoroid protection requirements appear to be the most significant environmental factor in the design of the astrionic module structure. A total skin thickness of 0.078 inch aluminum in a 1-inch sandwich construction is required to provide a 0.99 probability of no puncture for the 180-day quiescent mode (reference I-1, I-5). This requirement has a profound effect on structural weight as shown in Figure 4-6.

4.3.1.5 Weight

For a given set of structural requirements there exists a potential minimum weight design. Whether that minimum weight is really required is a decision that must be made as the result of trade studies. One of the prime elements is cost–and this cost must include a value for the worth of a payload increase. The weight budget adopted for the structure will then determine the degree of sophistication of the resulting structure. In general, severe weight limitations on the structure will result in more costly designs.

4.3.1.6 System Growth

System growth is one of the least obvious factors affecting the structural design. Figure 4-6 indicates that the required structural weight is relatively insensitive to payload weight, thus enhancing the view toward designing-in system growth by designing for the most stringent mission requirement.

4.3.1.7 Modular Interface

The tug structural design will be made compatible with the design of a standard interface docking mechanism that will be common to all space elements. Possible docking mechanism concepts include:

- "Pressure cooker" locking action–modules are mated and then rotated 5 to 10 degrees to insure positive locking
- Magnetic locking action
- Linear actuators with locking hooks
- Apollo type probe-drogue locking
- Pip-pin type locking (ball-in-groove)

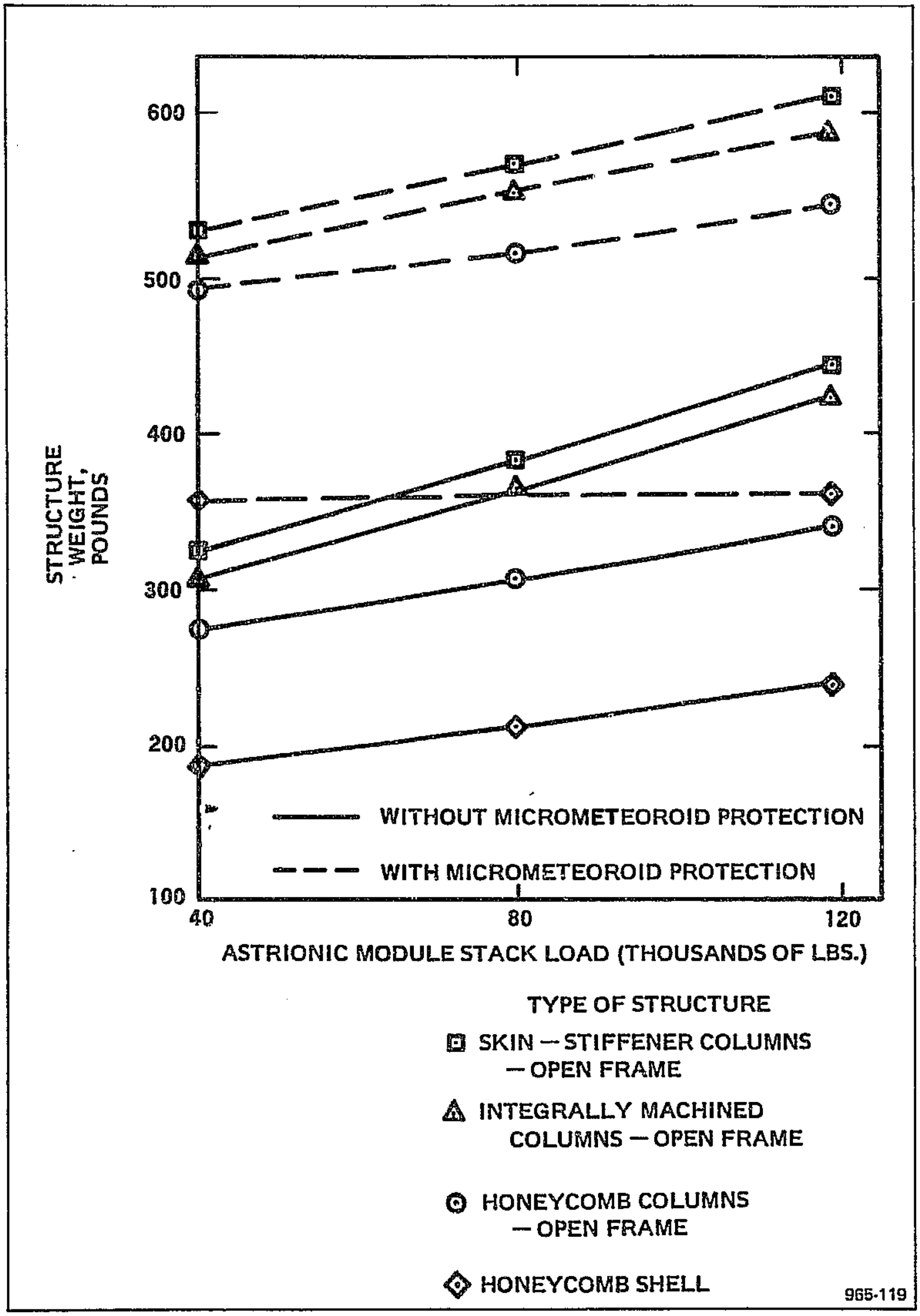


Figure 4-6. Earth Launch Environment Structural Loading

4.3.2 Design Concepts

4.3.2.1 Shell Structure

Research into the current literature on aerospace structures (references I-5 thru I-9) indicates that honeycomb sandwich structures offer the lightest possible structure available using current technology. The basic reason for this is that panels and cylinders loaded in compression may approach the material compressive yield stress--a feat not possible with any other design technique.

The material used for the study was 7178-T6 aluminum alloy face sheets and 5052 aluminum alloy core. These alloys were chosen as representative of similar alloys of ready availability.

Materials technology in the development of other materials for use in honeycomb sandwich construction has advanced significantly in recent years. As a result, it is now technically feasible, and in many cases cost-effective, to use as basic materials various combinations of

- Titanium
- Beryllium
- Boron Epoxy
- Carbon Epoxy
- Nylon
- Fiberglass

The baseline shell structure chosen for this study is considered to consist of:

- A basic shell of aluminum honeycomb sandwich
- Upper and lower interface rings
- Necessary brackets and fittings for equipment panel mounting
- "Mating" mechanism at the upper and lower interface rings

No allowance is made in the baseline structure for micrometeoroid protection. The following table is the weight breakdown for a "stack load" of 80,000 pounds:

Element	Weight (lbs)
Face Sheets (0.010")	48.8
Core (3.1#)	43.9
Upper Ring	19.7
Lower Ring	19.7
Brackets, Attachments	40.0
Mating Mechanism	40.0
	212.1

To provide adequate micrometeoroid protection, a face sheet thickness of 0.039 inches is required (reference I-1), and the total weight increases from 212 pounds to 355 pounds (see Figure 4-6).

The baseline structure weight was calculated for 40,000, 80,000 and 120,000 pound stack weights. These values define a curve of structural weight versus "stack load."

4.3.2.2 Open Frame Structure

This structure, shown in Figure 4-5, consists of eight columns configured to allow 48-inch wide panels to fit between adjacent columns. The configuration features upper and lower interface rings and a 6-inch deep band of honeycomb sandwich at the upper and lower interfaces to connect the eight columns together and provide a nominal torsional rigidity to the structure.

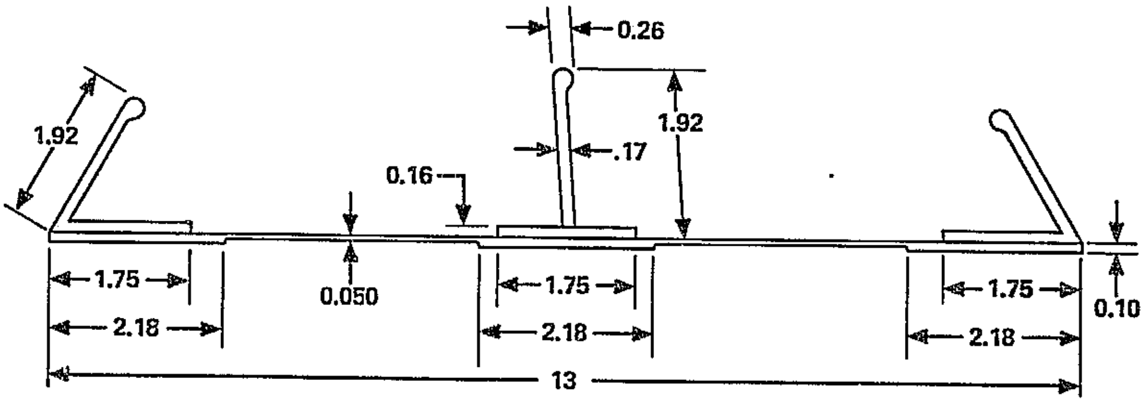
The main load-carrying element of the column is about 13 inches wide and runs the full depth of the module (48 inches). This element was designed as a stiffened-skin structure, an integrally machined panel structure, and as a honeycomb sandwich structure.

The side panels of each column are used as structure for the attachment of the mounting panels. The open design feature in this concept allows the options of either or both side component mounting or even hinging a panel for swing-out access.

The basic structure weights for these configurations are shown in Figure 4-6.

A brief discussion of the characteristics of the main load-carrying element (column) follows:

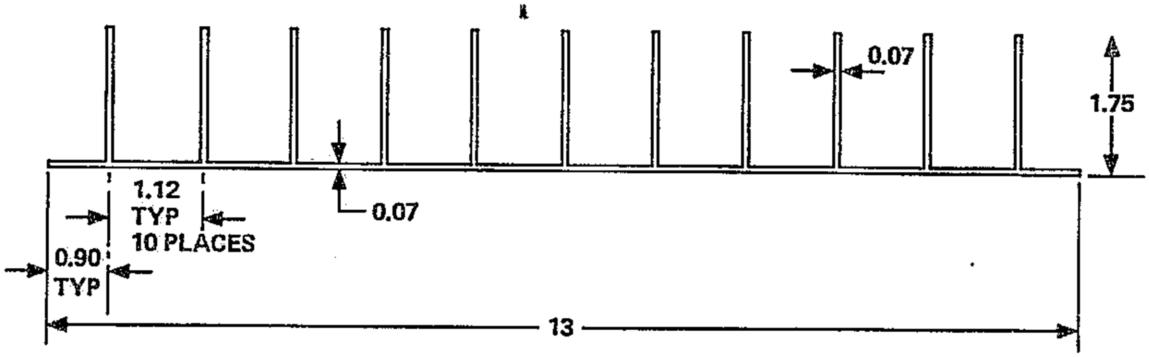
4.3.2.2.1 Stiffened Skin Construction. This configuration, View A in Figure 4-5, was designed as an element of a skin: one tee and two corner angles to join the side panels to the column. A total of 70 cross sections were investigated, and the least weight cross sections for 40,000, 80,000, and 120,000 pound stack weights were chosen. The cross section for 80,000 pound stack weight is presented below:



STIFFENED-SKIN COLUMN
80,000-POUND STACKLOAD

965-120

4.3.2.2.2 Integrally-Machined Stiffened Skin. This configuration, View B in Figure 4-5, is fabricated by machining a solid piece of material or an extruded shape to the required cross section. The following cross section is designed for a stack load of 80,000 pounds and represents a minimum weight cross section.



965-121

4.3.2.2.3 Honeycomb Sandwich. This configuration, View C in Figure 4-5, is of standard construction. Edge members are not shown. The weight breakdown for this configuration is that shown in Section 4.3.2.1.

4.3.2.3 Comparison of Shell and Open Frame Configurations

4.3.2.3.1 Shell.

Advantages:
- Least weight configuration
- Best stress distribution
- Micrometeoriod protection inherent in basic structure
- Thermal and environmental protection in basic structure

Disadvantages:
- Accessibility limited by basic configuration
- Local reinforcement required for every load concentration point

4.3.2.3.2 Open Frame.

Advantages:
- Maximum accessibility to components
- Maximum flexibility of panel mounting schemes

Disadvantages:
- Less efficient structurally
- Heavier weight than shell
- Requires additional means of micrometeoroid protection
- Requires additional means of thermal and environmental protection

4.4 MODULE ASSEMBLY

4.4.1 Access

Figures 4-7 and 4-8 show the general layout of the assembled module for the shell structure and open frame structure, respectively.

As can be seen in Figure 4-7, access to the components for maintenance operations for the shell structure would be through a hinged access opening which also serves as a part of the extract/retract mechanism for the high-gain antenna and landing radar. Equipment and the radiators are mounted to the interior and exterior of the shell, respectively, by mounting pads. Fluid lines and cabling are routed through brackets attached to the structure above and below the panels. For these operations, internal lighting would be required. In the event of major AM reconfiguration, the AM could be destaged from the forward module to provide increased accessibility.

The radiator and louvers are not shown in the overall view of Figure 4-7 for clarity; however, they are depicted in the sectional view.

Figure 4-8 shows the accessibility advantage of the open frame structural concept. Access to the components is from the outside through individual radiator doors hinged from the vertical columns. The component mounting panels are mounted to clips located on the inboard edge of the triangular vertical columns. Figures 4-9 thru 4-13 show several alternate component mounting and access methods.

Figure 4-9 shows the expansion capability of the open frame structural concept to accept additional equipment. By omitting one panel, the unit may be entered through the radiator/door, and components may be mounted on the internal surfaces of the seven remaining panels. Inside access could also be obtained by removing the module immediately forward of the astrionic module (i.e., crew or cargo module).

Figure 4-10 shows the capability of using both sides of the mounting panels by hinging the panel in two places. This provides external or internal access to both sides of the panels as well as doubling the available surface area for mounting equipment.

Figure 4-11 shows the cover plate/radiators and the component mounting panels mounted in a sliding bracket. Access is attained by pulling the cover plate/radiators outboard and exposing both sides of the component panels. A folding seat/platform may be used to hold the crew member in position while working.

Figure 4-12 shows the expansion capability of the shell structure to accept additional equipment. Hinged brackets holding the panels inboard will allow component mounting on both sides.

Figure 4-13 shows the possibility of access to all components by opening two clamshell doors. This concept requires a smaller structural ring. Expansion is limited to added height.

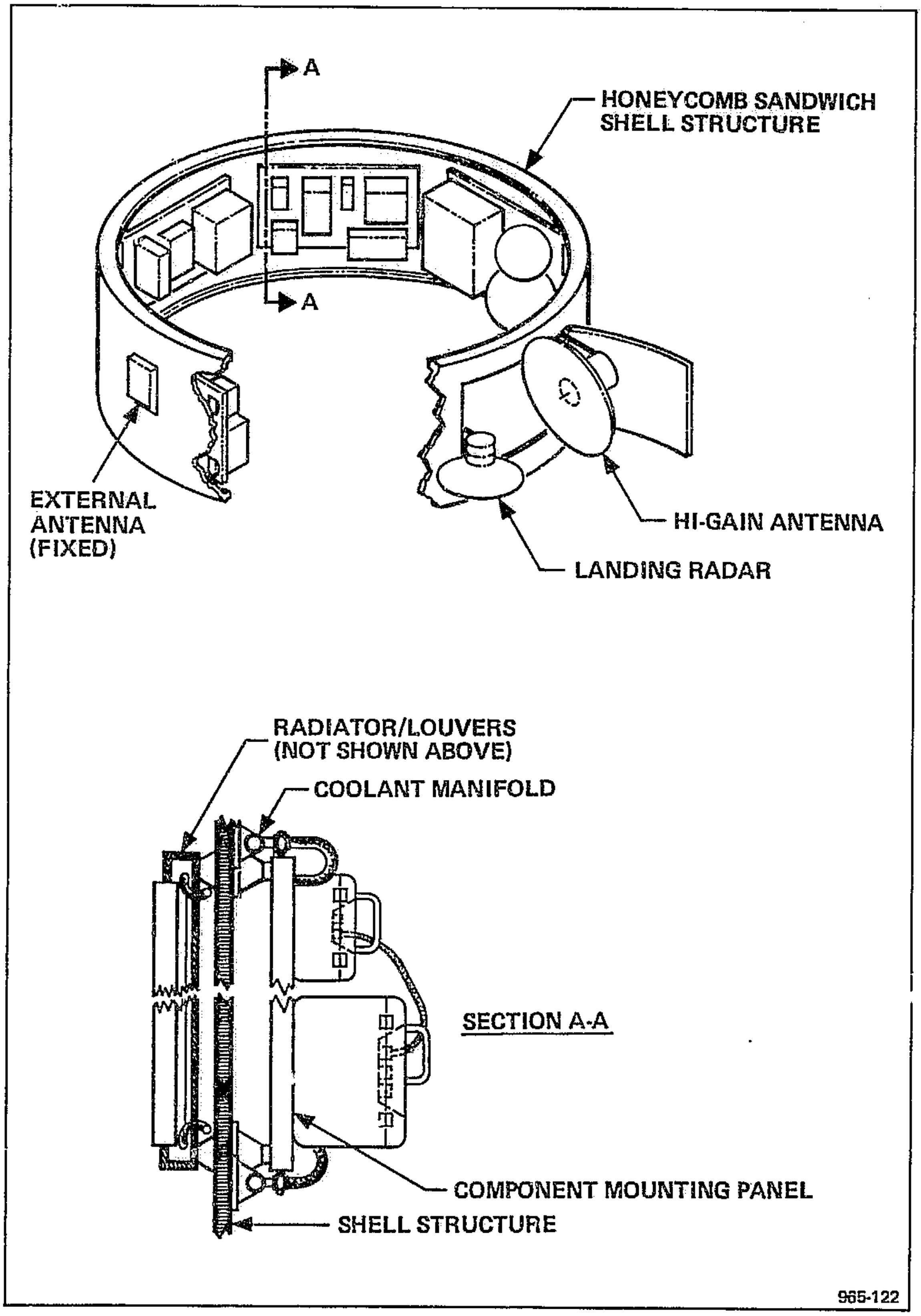


Figure 4-7. Astrionic Module Shell Structure

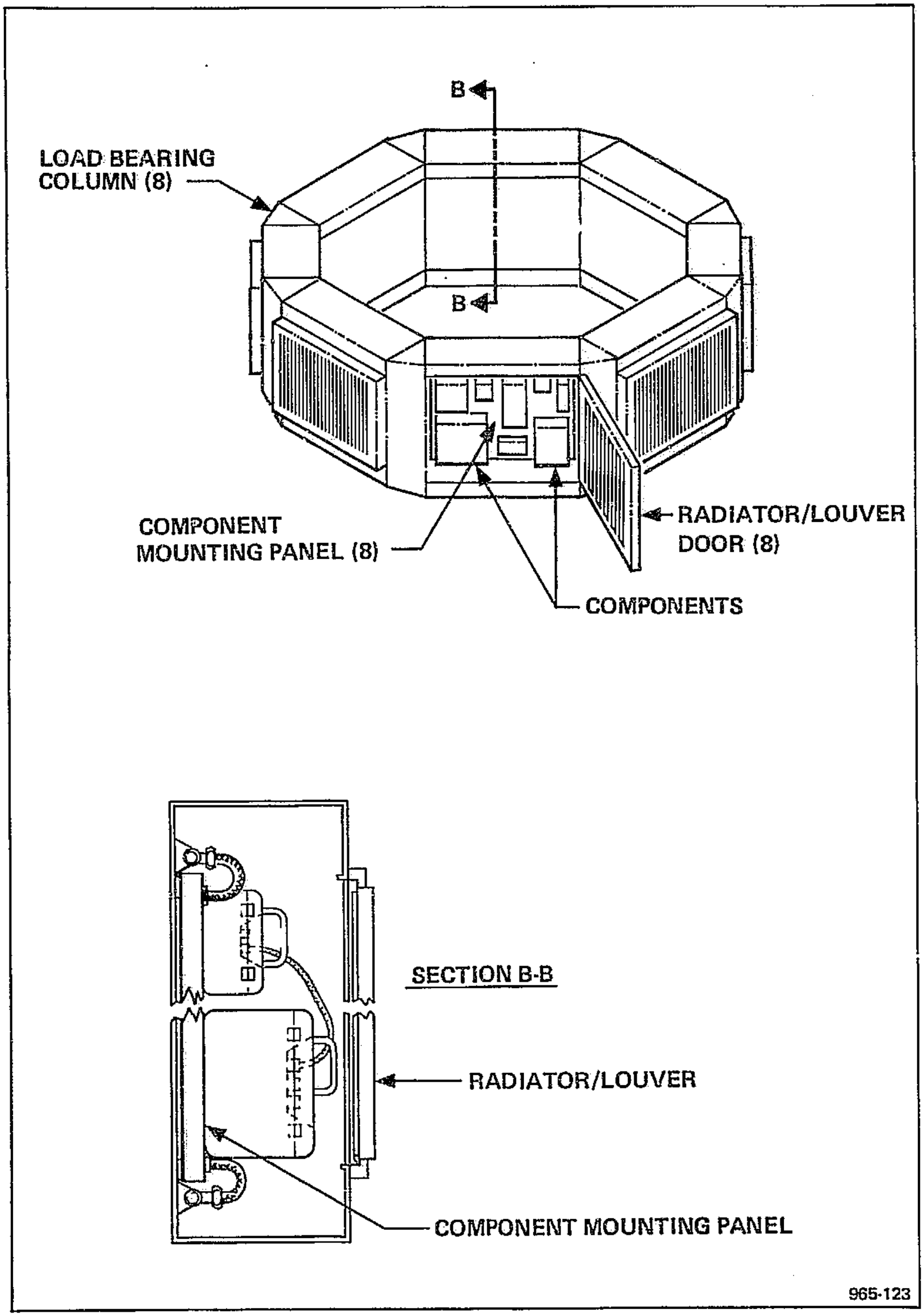


Figure 4-8. Astrionic Module Open Frame Structure

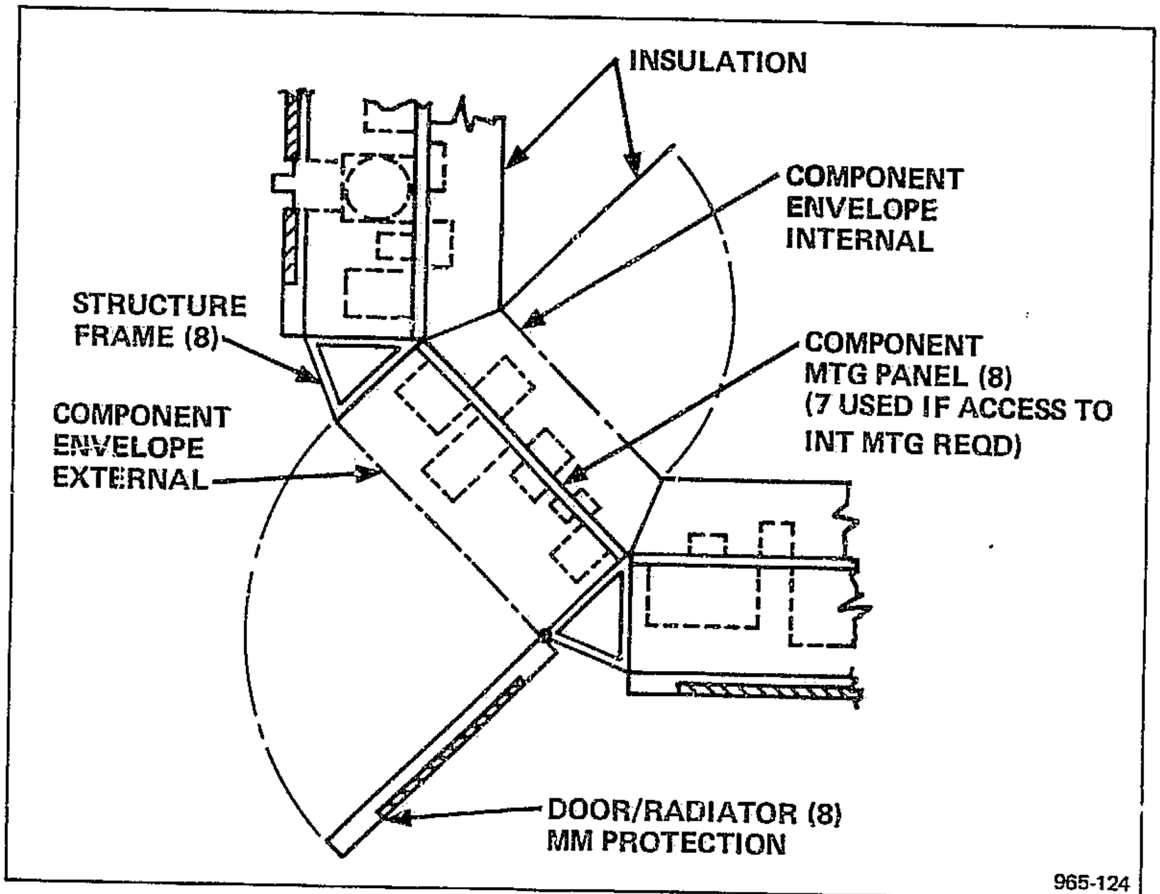


Figure 4-9. Open Frame Structural Option Alternate Access

In summary, there are many feasible concepts of mounting the component panels to the basic open frame structure. Selection of the optimum design should be made in follow-on studies when accessibility needs are better defined. It was shown that a single concept can be devised which would allow either internal or external access, if necessary.

4.4.2 Antennas

Antennas for the USB and VHF systems shall be hard-mounted to the external skin or frame.

The high-gain antenna and the landing radar antenna, when required, shall be mounted on swing arm type frames and will be stored inboard during the powered down phases. Upon the signal to activate, these antennas will swing out to operating position and gimbal, as required.

4.4.3 Viewports

Certain tracking devices require viewports to the outside of the structure. These ports will be held to a minimum size to maintain structural adequacy. Micrometeoroid protection for these devices must either be self-contained or a retractable micrometeoroid cover provided.

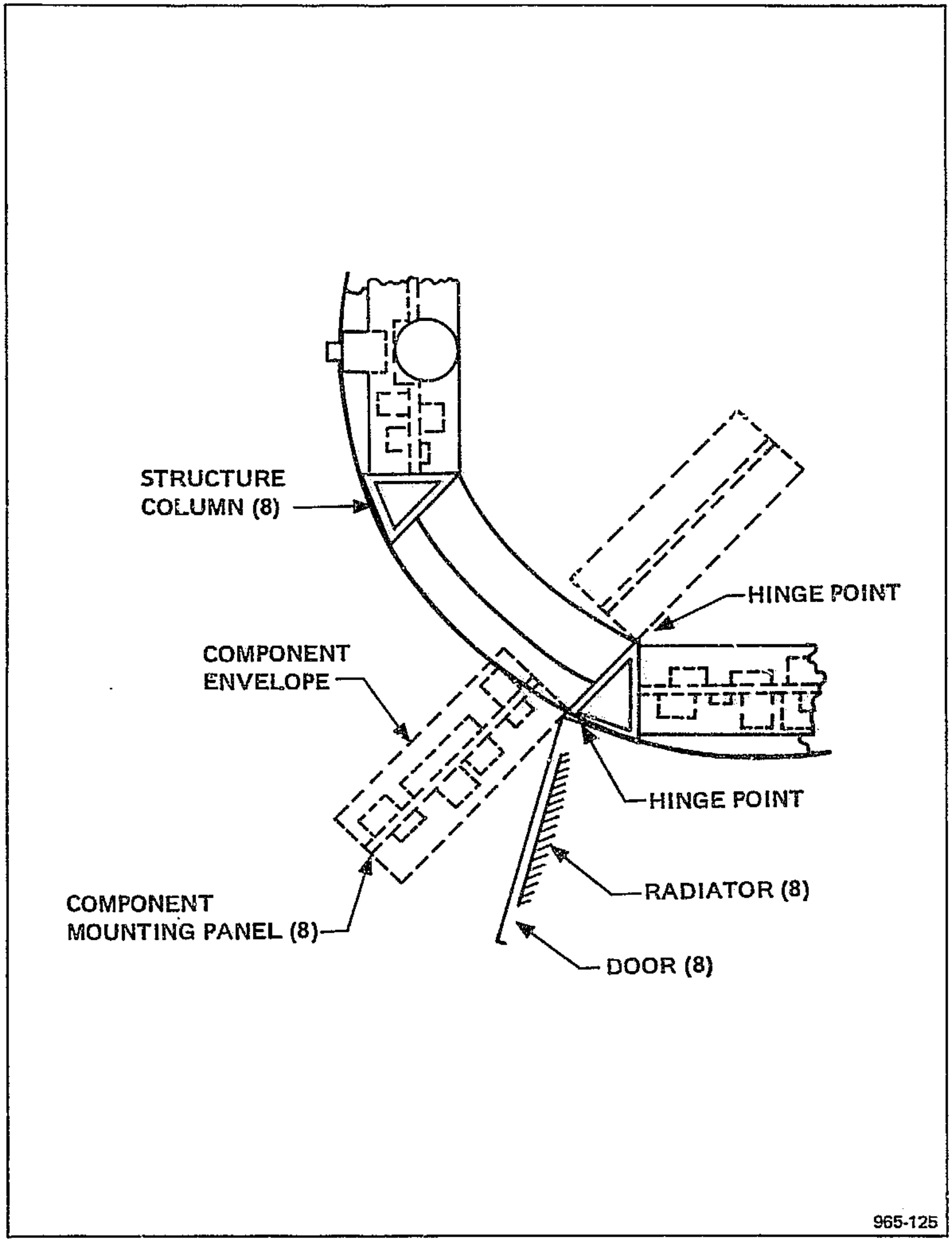


Figure 4-10. Maximum Accessibility Open Frame Structure

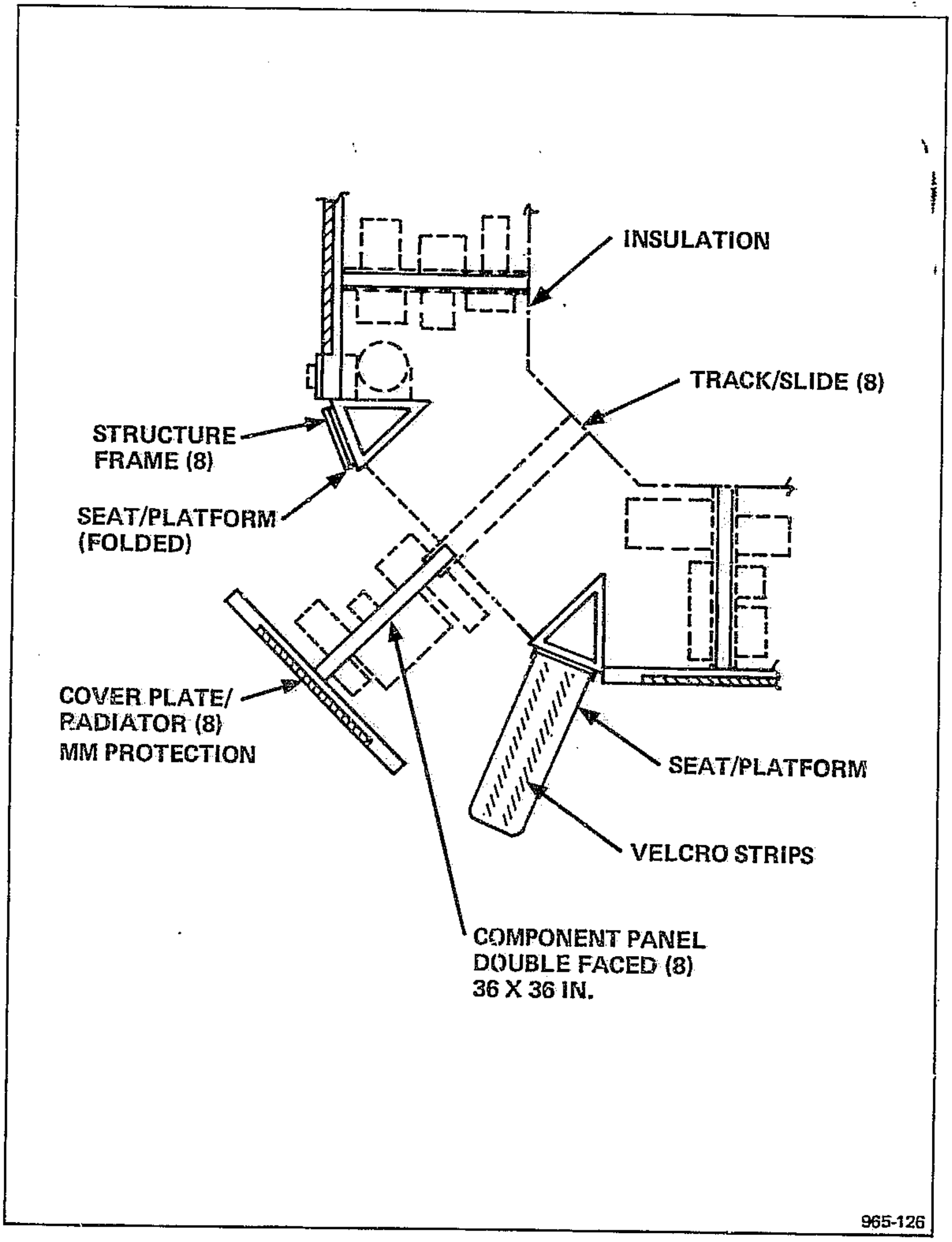


Figure 4-11. Open Frame Structural Option Alternate Access

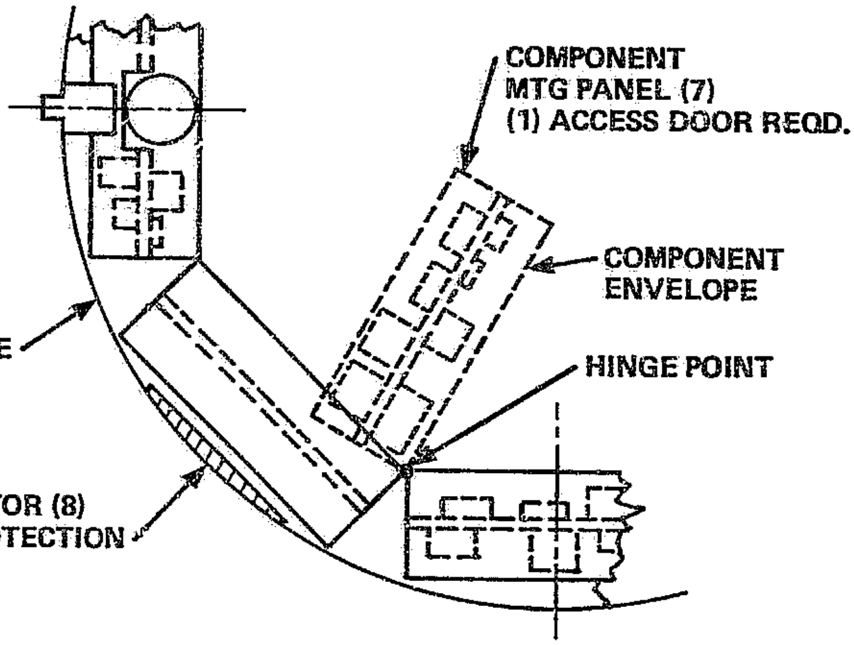


Figure 4-12. Shell Structure Option Alternate Access

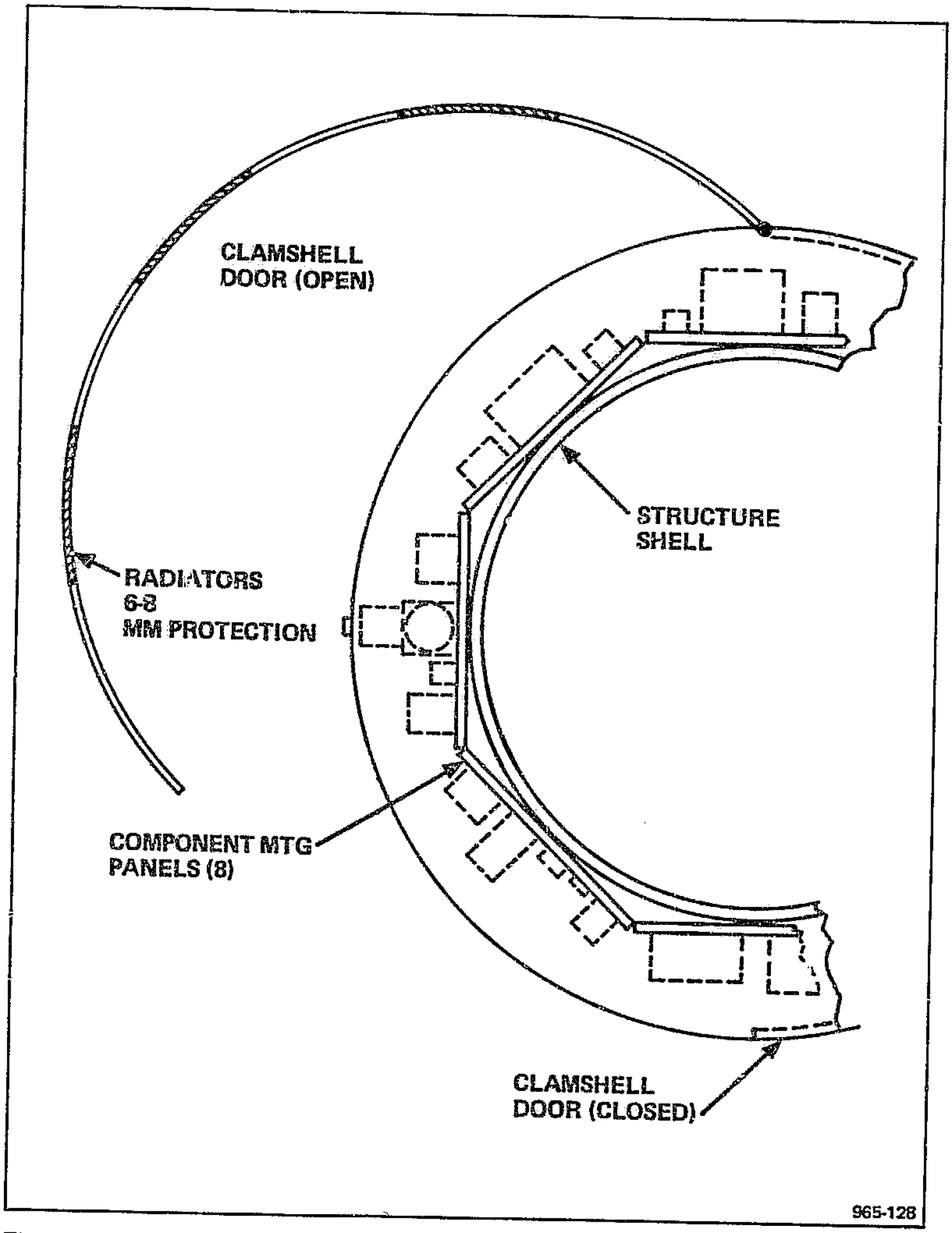


Figure 4-13. Shell Structure With Clamshell Door External Access

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APPENDIX J

SPACE TUG THERMAL CONTROL ANALYSIS AND IMPLEMENTATION

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

This study is directed toward defining the thermal control problem and identifying candidate thermal control concepts for the space tug astrionic module application. The investigation first discusses the projected space tug missions and categorizes them according to common phases of thermal environment. Passive thermal control techniques are then considered in an attempt to define a passive system for maintaining temperature control in these environments during the extended dormant (storage) mission phases. Active conditioning techniques are investigated for those phases of the mission requiring additional thermal control capability.

The system recommended as a result of this brief investigation combines passive and active conditioning methods and is flexible toward the multi-mission, reusable system role of the space tug vehicle.

2.0 STUDY GUIDELINES AND GROUNDRULES

Overall guidelines, groundrules and assumptions for the space tug study are discussed in previous appendices of this report. Details and/or additions which particularly influence the thermal control study are presented in this section:

- Thermal control will be provided for only the astrionic module
- The astrionic module will be nominally 13½ feet in diameter and height will be optimized
- The astrionic module will be packaged to allow a remove-and-replace maintenance and reconfiguration concept in space
- The tug shall be reusable by refueling, replacement of consumables and minimum refurbishment
- External power and environmental conditioning may be provided to the space tug astrionic module by a space element when the tug is docked with the element
- The space tug shall be compatible with the earth and lunar orbit space stations, the space shuttle and the reusable nuclear shuttle
- The space tug will be delivered to orbit by a space shuttle or Saturn derivative vehicle
- The space tug shall be capable of maintaining a quiescent status for up to 180 days in earth or lunar orbit when docked to other vehicles or free flying.
- The low earth orbit is defined as between 100 and 280 nautical miles altitude with a range of inclination of 28.5° to 55°.

Thermal conditioning is required to maintain an acceptable temperature environment for the astrionic equipment. Primary factors in achieving this end are the allowable temperature range of the equipment, the electronic heat to be dissipated and the thermal environment in which the equipment must operate. The former two are dependent upon the specific electronic components comprising the astrionic system. Based on the identified astrionic equipment list and component type characteristics, allowable temperature limits of 0-130°F for inactive periods and 45-100°F for active periods appears reasonable. Maximum component power dissipation was determined to be on the order of 1000 watts for all missions. In addition, 300 watts cooling capability will be required for conditioning the two kilowatt fuel cell of the electrical power system. Preliminary duty cycle estimates indicate that the continuous (average) level of power dissipation is sufficiently close to the maximum level that the latter may be used in determining total requirements for the active mission duration. Thus, the thermal control system will be sized to accommodate 1.3 kilowatts of electronic power dissipation.

The thermal environment is unique for each vehicle mission, as it is dependent on flight path and vehicle attitude, and also varies with time of year and day of vehicle launch. Since most of these conditions are not known at this phase of the investigation, worst case thermal environments were assumed in thermal control considerations.

- Cold Case – Minimum heating conditions occur for module with no exposure to solar radiation or for vehicle in maximum shadow orbit

- Hot Case – Maximum heating conditions occur for module continuously exposed to solar radiation or for vehicle in minimum shadow orbit.

In addition, it is assumed that the astrionic module is completely isolated, thermally, from all other stages and/or systems.

The mission description for the space tug vehicle reveals, as regards the external thermal environment, that the broad spectrum of missions can be categorized (with only one exception) to include essentially three common mission phases:

- Low Earth Orbit (LEO) – Each mission begins with the space tug vehicle linked with a space element in low earth orbit. Orbit parameters are variable and include inclinations between 28.5 and 55° and altitudes from 100 to 280 nautical miles. Astrionic systems are powered up to begin the vehicle mission. This initial phase ranges up to 5 hours in duration (with systems powered up).

- Transfer Phase – The vehicle initiates a transfer trajectory toward a specific objective including synchronous orbit, translunar and deep space coast. This mission phase ranges up to 57 hours active system duration.

- Low Earth Orbit – The vehicle returns to low earth orbit and goes into a storage mode prior to refurbishment for the next mission. This storage mode may be either docked with another space element or free-flying and last up to 180 days. The orbit parameters may or may not be identical to its initial orbit phase.

The only exception to the above generalization occurs in the case of the lunar mission. Missions which include a lunar orbit or lunar surface operation are distinctly different from all other missions and present unique thermal environment problems, the technical depth of which put them beyond the time scope of this study. (A brief discussion of some of the thermal aspects of the lunar missions is given in Section 4.4.)

All remaining thermal control considerations will be directed toward satisfying the thermal environment described by the above paragraphs.

3.0 SUMMARY OF RESULTS

The preliminary analyses of this investigation indicate that thermal conditioning of the space tug astrionic module can be achieved with present state-of-the-art techniques while maintaining flexibility toward the multi-mission role.

The extended duration of the dormant (storage) mission phase suggests that passive type thermal control methods be utilized. The additional capability required during the active mission phases demands an active thermal control system.

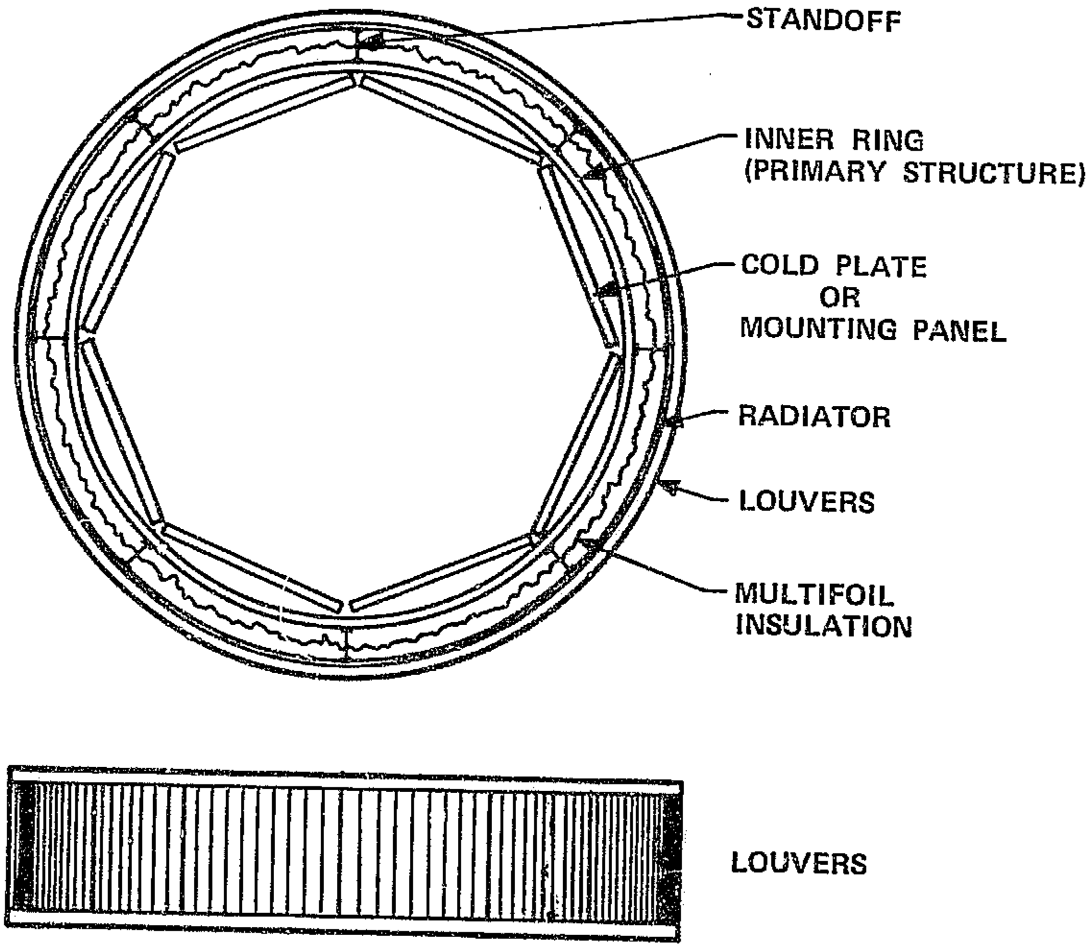
- Passive control is possible through a high degree of thermal isolation between the vehicle internal compartment and the external environment. This is achieved in the recommended system by locating all major astrionic components on an inner structure which is surrounded on the outboard side by an outer shell. Highly effective multi-foil insulation and the use of low conductive structural materials between the two elements provide the necessary isolation. Thermal control coatings on the outer shell provide the necessary interface with the external radiation environment.

- Active control is achieved through a circulating fluid, closed loop system. The electronic packages which require cooling are mounted on coldplates which are located on the inner structural element. The primary trade item was the type of heat sink device to be employed. The prime candidates were expendable fluid heat rejection systems and space radiators, which require no expendables. The radiator proved to be the optimum choice for the spectrum of space tug missions. Radiator size requirements suggest that the radiators form the outer shell of the astrionic module. To integrate the active and passive systems through this common link, the use of temperature driven mechanical louvers is necessary to cover the radiator surface and provide the proper surface area/optical property control. Thus the "passive system" described in the above paragraph is more accurately termed semi-passive.

The thermal control concept recommended as a result of this study is the same for all projected missions, with the exception of the lunar missions, and is depicted in Figures 4-1 and 4-2. As described in previous paragraphs (Section 2.0 above), the requirements of the lunar missions were not considered in the selection of the recommended system.

4.0 DETAILED ANALYSIS

The space tug mission description reveals that the greatest portion of the space tug mission is spent in the quiescent or dormant phase in which a predominant portion of the astrionic equipment is either powered down or used only infrequently. As this period is extensive (up to 180 days) when compared with the active portion of the mission (up to 60 hours), the approach taken in the analysis was to first consider the dormant period then expanding and/or refining the concepts to include the entire mission spectrum. The lunar missions are distinctly different from the other tug missions and will be detailed in Section 4.4.



NOTE: THIS FIGURE ILLUSTRATES A THERMAL DESIGN CONCEPT ONLY. DETAIL STRUCTURAL BREAKDOWN IS SHOWN IN APPENDIX I.

965-129

Figure 4-1. Passive Thermal Control Structure

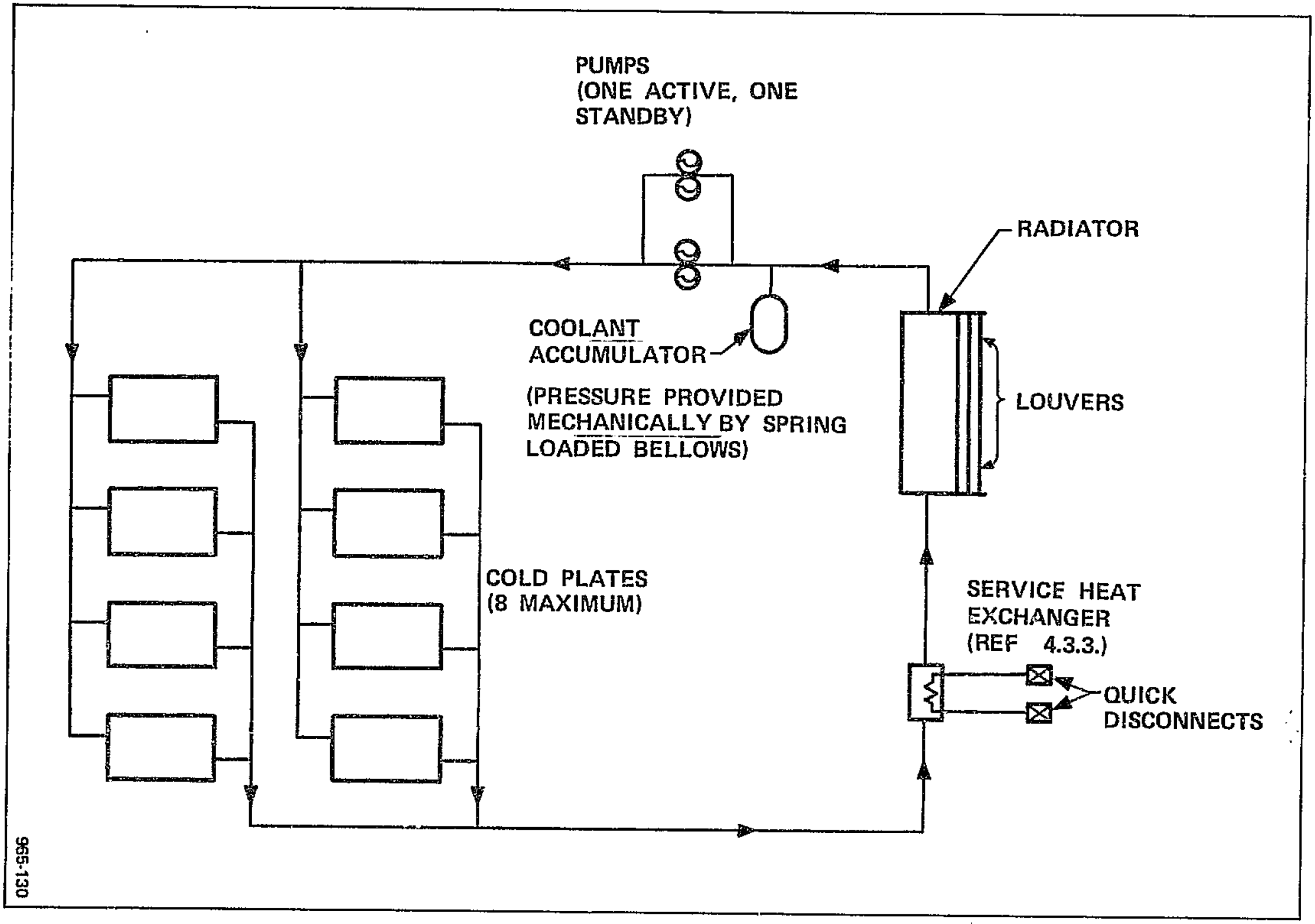


Figure 4-2. Active Thermal Control System Schematic

4.1 DORMANT/QUIESCENT MISSION PHASE (EARTH ORBIT)

The duration of the dormant mission phase dictates that prime emphasis be placed on passive-type thermal control concepts. Assuming a general astrionic module structural configuration as shown in Figure 4-1 with the electronic components located on or within the inner structure, this becomes largely a problem of determining the required optical properties of the outer skin and defining the thermal link between components and outer skin which will insure that the components will remain within allowable temperature limits. Design for thermal control using passive techniques such as this require exact knowledge of vehicle attitude, particularly relative to solar radiation. Presently this is not defined, and it is not possible to determine exact surface requirements. However, it is beneficial to consider some of the known "worst case" conditions and assess the overall feasibility of the passive control approach.

The maximum and minimum low earth orbit heating conditions for the vehicle occur as follows:

- Maximum Vehicle Heating – 280 n.m., 55° inclination orbit with the vehicle continuously exposed to the sun
- Minimum Vehicle Heating – 100 n.m., 28.5° inclination orbit (maximum time in earth's shadow)

With the aid of a computer program developed to study the thermal environment and temperature of orbiting space vehicles (Reference J-1) the incident heat loads were determined for an earth/velocity oriented vehicle. The results are shown in Figures 4-3 and 4-4 for the above maximum and minimum heating cases, respectively. Total incident heat, solar + albedo + earth infrared, is shown as a function of time (position in orbit) for the quadrant vehicle locations. Average values over the total vehicle surface are approximately 175 btu/hr/ft^2 and 90 btu/hr/ft^2 for the maximum and minimum heating cases, respectively. Using the average incident heat with the simplification that the outer shell temperature is constant around the shell and has an allowable range equivalent to that of the components (0-130°F), the required optical property ratio(α_S/ϵ) was determined by an energy balance. The results, presented in Figure 4-5, show that to satisfy the maximum heating case $\alpha_S/\epsilon \leq 1.2$; similarly the cold case demands $\alpha_S/\epsilon \geq 0.8$. Thus a thermal control coating with a solar absorptivity/emissivity ratio between 0.8 and 1.2 would be sufficient to maintain passive temperature control. These values are well within the range of that attainable with existing coatings.

The assumption of constant temperature and incident heat around the vehicle is equivalent to assuming vehicle rotation such that an even temperature is maintained. While not being strictly true, this is sufficient for a first approximation. The utilization of super insulation between the outer shell and the internal compartment, in addition to dampening the effect of the highly transient orbital thermal environment, significantly increases lateral conduction around the vehicle outer shell. It should be noted that the above results are independent of the total mission time in the dormant (orbit) phase.

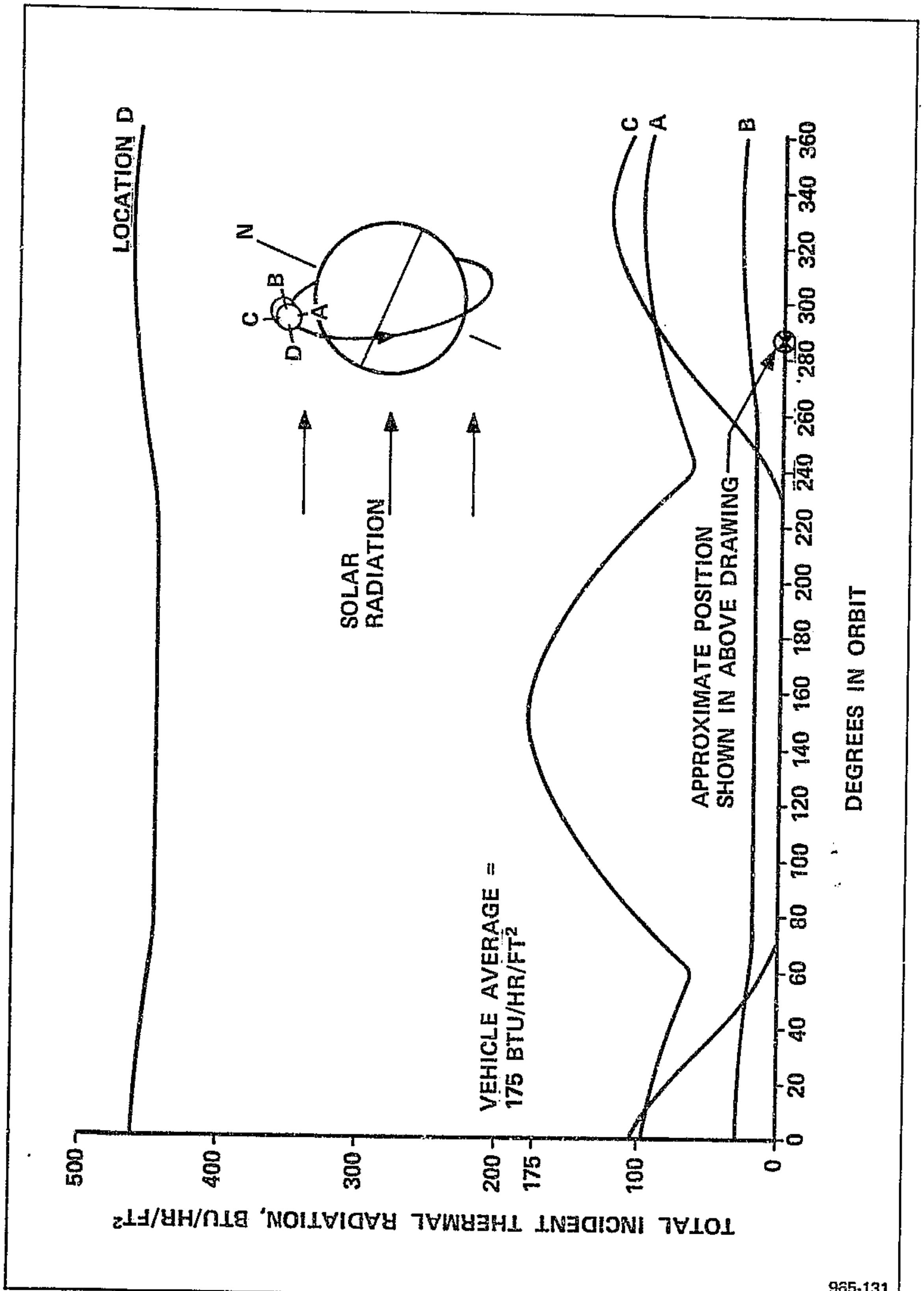


Figure 4-3. Incident Heat VS. Position In Orbit Maximum Heating Case (No Shadow) 280 NM 55° Inclination LEO

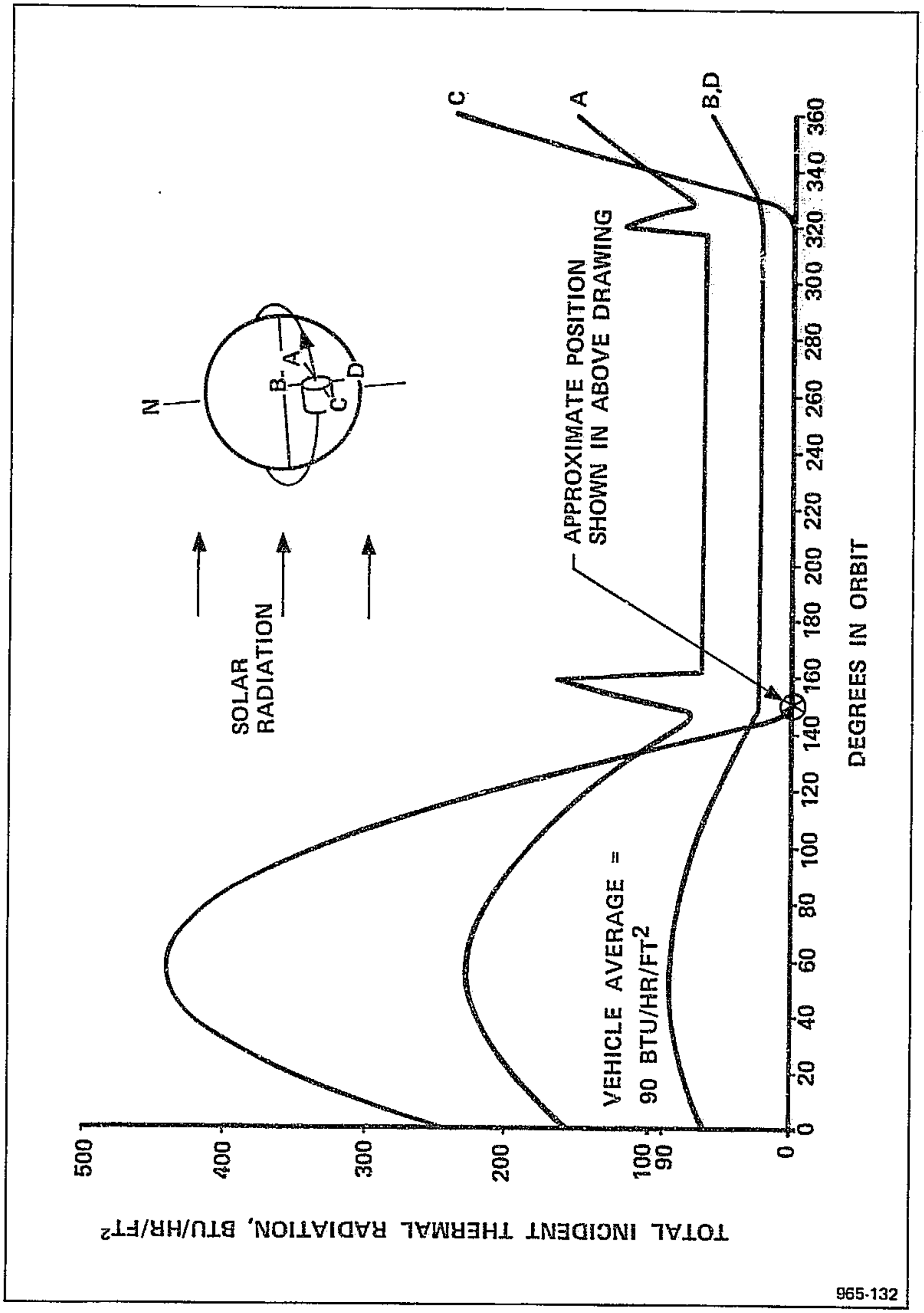


Figure 4-4. Incident Heat VS. Position In Orbit Minimum Heating Case (Maximum Shadow) 100 NM 28.5° Inclination LEO

Some orbital heating conditions are possible in which the range of the thermal environment is such that no one set of optical properties on the outer skin will suffice for both maximum and minimum extremes. (This is not the case for missions considered in this study; however, it probably will exist for the lunar missions to be studied later.) In the hot case, a low α_S/ϵ ratio is desired; in the cold case, the opposite is true. Under such conditions, the only way to maintain completely passive thermal control is to impose a constraint on vehicle attitude. In this manner, the thermal environment range to which the vehicle is exposed is essentially narrowed (controlled) such that one set of optical properties will suffice. This, however, may not be a feasible approach for the space tug mission, particularly in light of the long duration of the dormant mission phase. In this case, an alternate method of maintaining dormant phase temperature control with other than an active system is to employ a technique such as that shown in Figure 4-6. This system, referred to as semi-passive, controls the radiation heat exchange by providing two surfaces with different optical properties and varying the percent area of exposure of the two surfaces via a system of louvers. The position of the louvers is controlled by means of temperature sensitive bi-metallic thermal actuators. In this manner, the optical properties are effectively changed with the varying thermal environment. Systems of this type have been used or proposed on a number of space vehicles including Mariner, OGO, Pioneer, Pegasus and Voyager.

Figure 4-7 illustrates a typical case of the effect that louvers would have on surface temperatures. The temperature profiles shown are for measurement point "A" in Figure 4-4. The solid line represents the temperature of a surface with fixed optical properties, and the dashed line represents the temperature of this surface with two sets of optical properties (louver system). As shown, peak temperatures are significantly less with the louvers. The dampening effect could also be achieved on the minimum temperatures by using a higher α_S/ϵ value for the louver system (e.g., $\alpha_S/\epsilon = 0.9/0.75 = 1.2$ instead of $0.9/0.9 = 1.0$).

It will be seen in later paragraphs that one of the prime candidates for the heat exchange device for active thermal control is a space radiator. The effectiveness of such a device is strongly dependent upon the absorptivity/emissivity ratio of its radiating surface. A low α high ϵ combination is naturally desired to minimize the effect of incident thermal radiation and maximize the emitted energy. That such a device (low α_S/ϵ) is incompatible with the optical property values determined above ($0.8 \leq \alpha_S/\epsilon \leq 1.2$) for the passive system can be offset by employing a louver system such as that shown in Figure 4-6. In this figure, the low α_S/ϵ surface would constitute the external surface of the space radiator.

The conclusions to be drawn at this point are that while exact vehicle attitudes are not fixed, a first approximation using existing mission parameters and known worst case conditions of inclination and attitude indicates that passive thermal control is feasible for the dormant/quiescent mission phases. However, to properly integrate with the active thermal control concept will require the use of semi-passive techniques (louvers).

4.2 ACTIVE MISSION PHASE

In the active phases of the mission, two additional parameters enter into the thermal control problem:

- Component temperature requirements are generally more stringent in the active phase.
- The components dissipate electrical power in the form of waste heat.

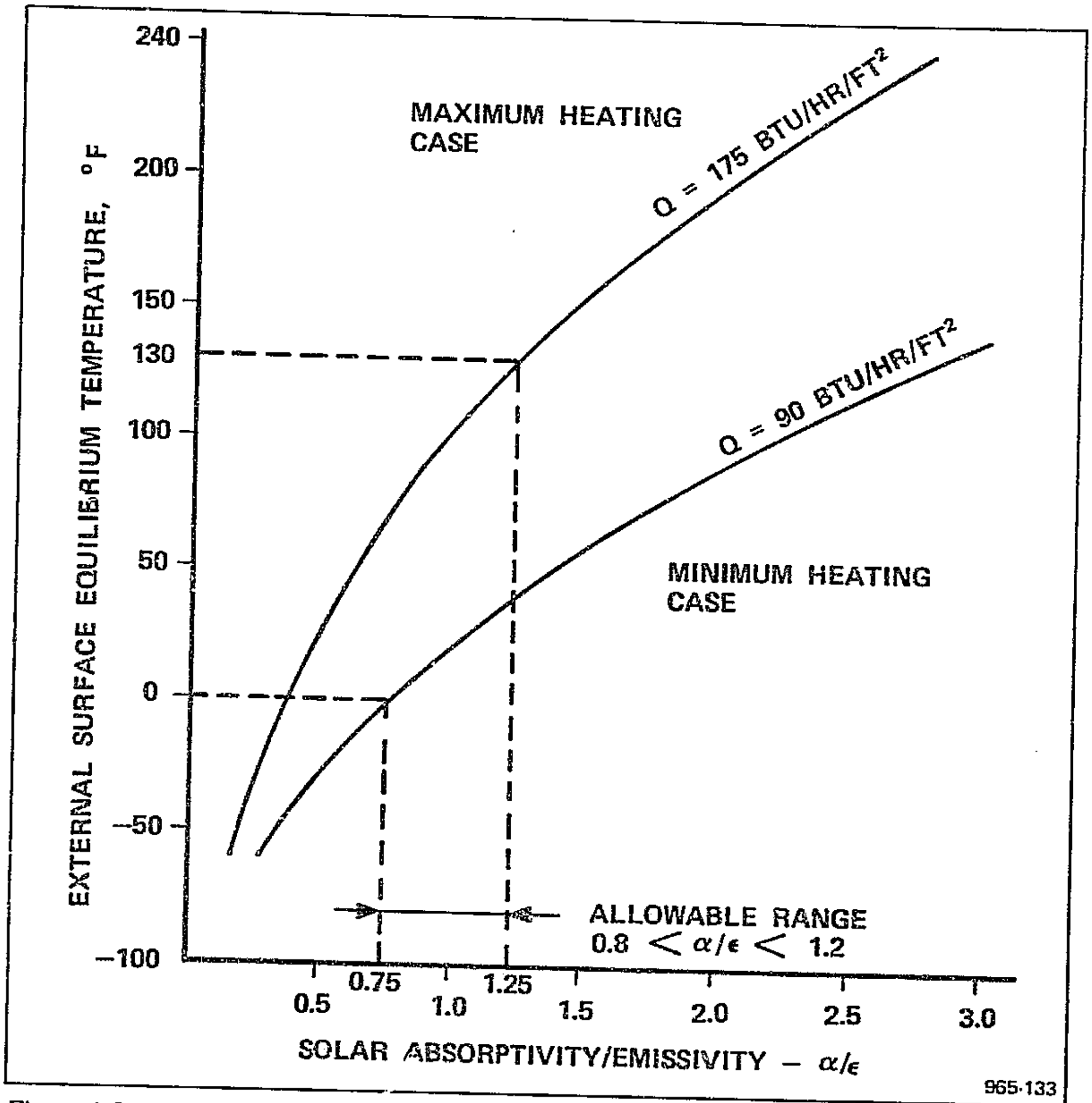


Figure 4-5. Allowable Range of α/ϵ for Passive Control

These parameters are physically opposed and dictate a firm requirement for cooling. To cool passively would require, at a minimum, an intimate thermal link with the external environment. However, in order to maintain a passive type system for the long term dormant mission phase, the design evolved thus far in this study has resulted in a system which is thermally isolated from the external environment. Thus while passive techniques are sufficient to maintain the astrionic components within allowable temperature levels in the dormant phases of the mission, such is not the case when the components are powered up. For the active phases of the mission, active thermal control is required to supplement the passive methods and provide the additional capability required for these periods. Figure 4-7A indicates this graphically (Reference J-3).

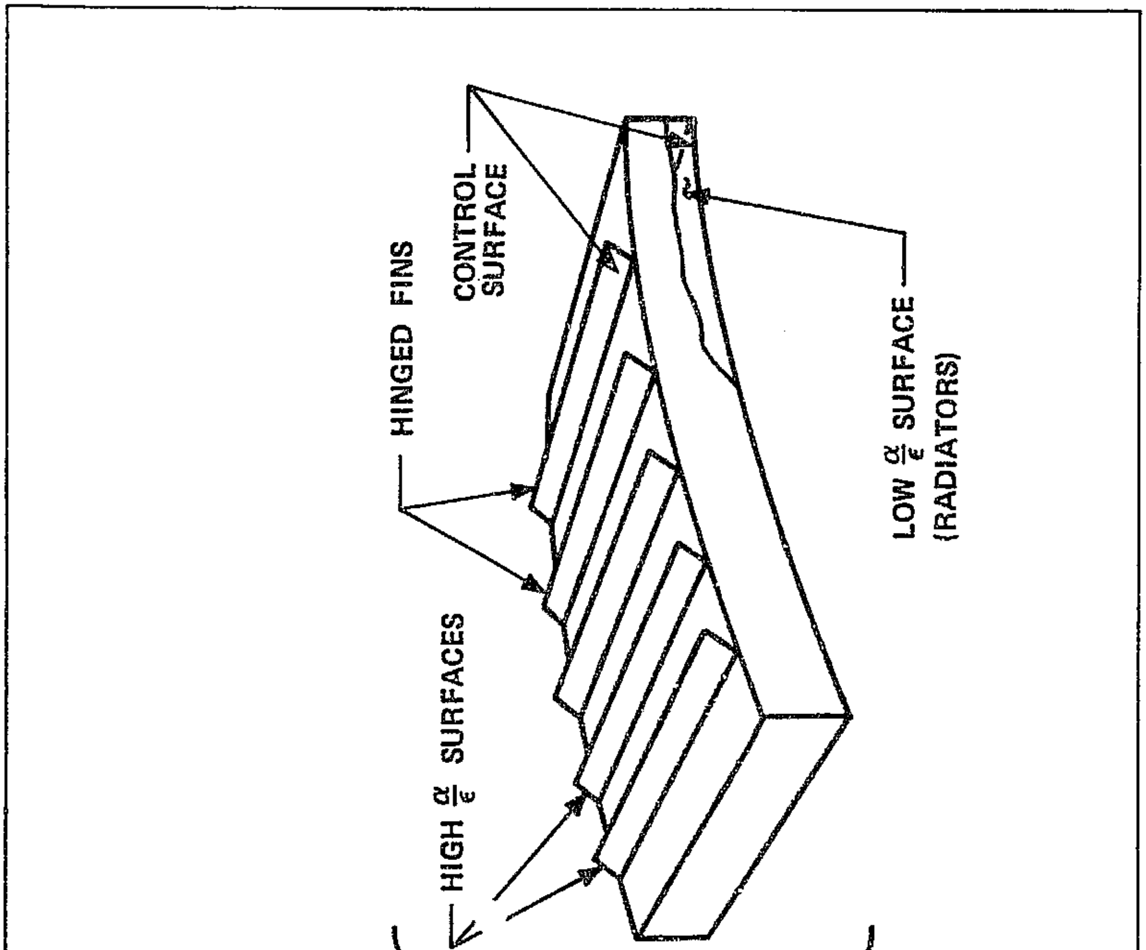


Figure 4-6. Louver System Concept

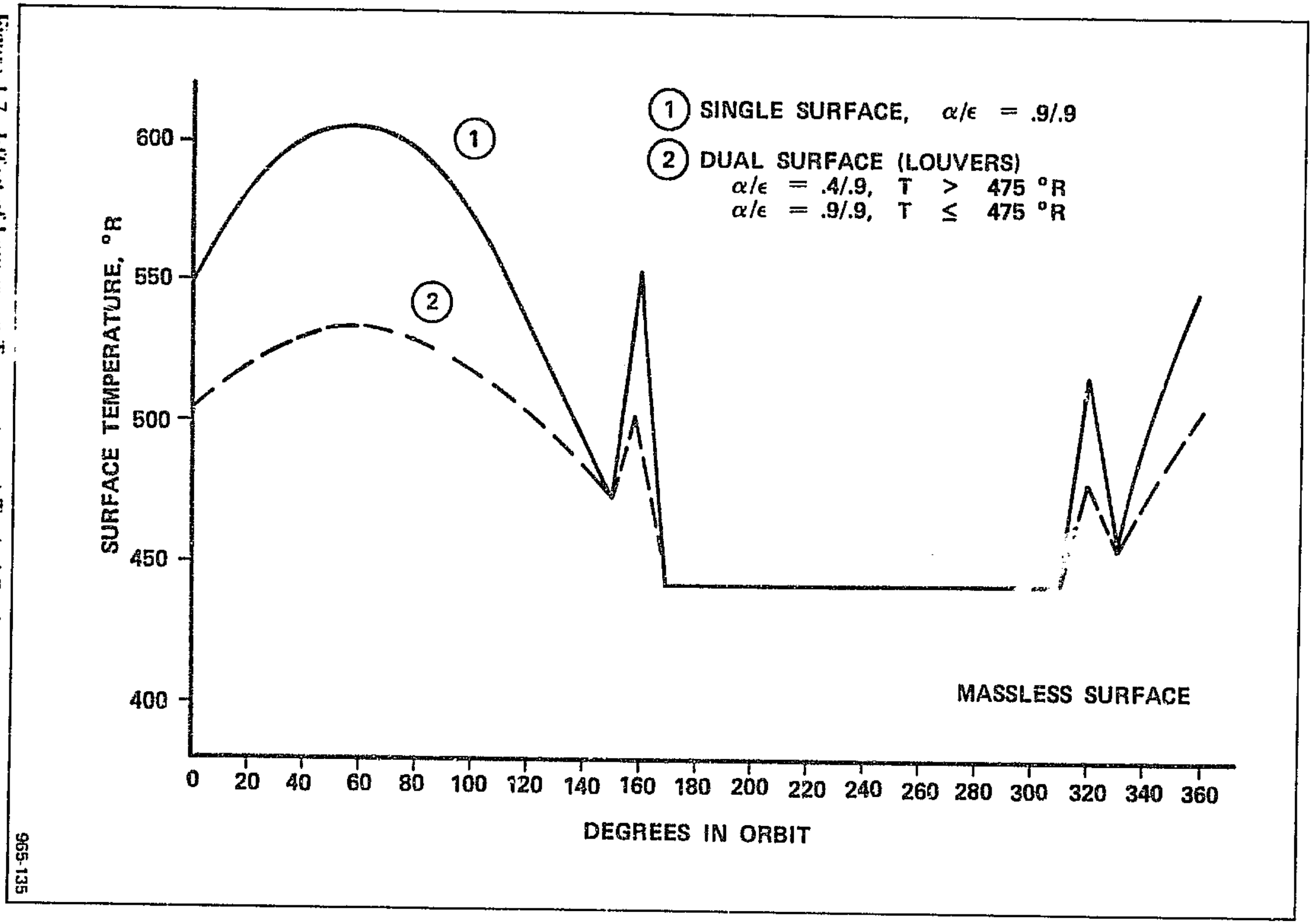
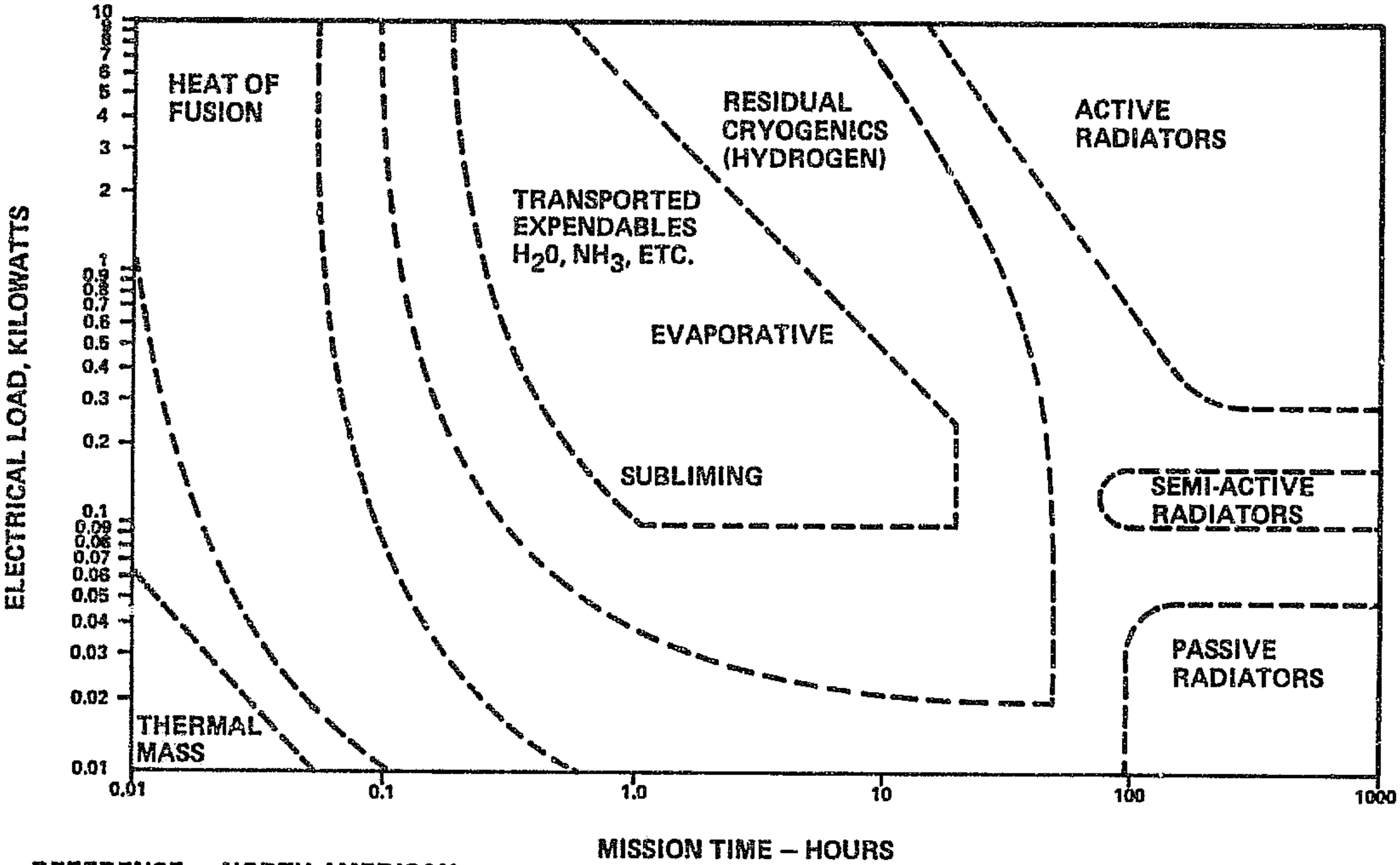


Figure 4-7. Effect of Louvers on Temperature of Typical Surface



REFERENCE: NORTH AMERICAN ROCKWELL, INC.

965-136

Figure 4-7A. Regions of Applicability for Heat Sinks

The optimum thermal design for the astrionic module application is to employ a closed loop fluid circulation system such as that shown schematically in Figure 4-8. The electronic packages are mounted on thermal conditioning panels (coldplates) through which the coolant fluid circulates. Heat is transferred to the fluid by conduction and convection. The fluid then rejects the heat from the system at a central heat sink. Closed loop systems such as this are highly flexible with varying mission requirements and provide control with relatively narrow temperature control bands. The heat sink selected for use on a particular system is dependent on a number of factors including the amount of heat to be dissipated, degree of temperature control required and operating life as well as weight, cost and reliability. For the realm of operation of the space tug astrionic module, the optimum systems are those employing either expendable fluid heat rejection systems or space radiators.

Expendable fluid systems are those which utilize the latent heat associated with a change of phase of a secondary fluid. Two prominent types which have been used to varying degrees in a number of spacecraft applications are water boilers and sublimators. General performance of water boilers is less satisfactory than sublimators, particularly with fluctuating heat loads. Control problems are inherent in the water boiler system, and in addition, weight requirements are generally in excess of those for the sublimator system. For these reasons, water boilers were eliminated from further consideration.

Thus, the active system heat exchanger options have been reduced to two: a sublimator or space radiator. The following paragraphs will discuss how each option would relate to thermal control of the astrionic module and identify pertinent trade items to be considered in selecting the optimum configuration.

4.2.1 Space Radiator

Simply stated, a space radiator rejects heat directly to deep space by virtue of the temperature and optical properties of its radiating area. In the space tug astrionic module application, the radiator(s) would be integrated into the outer shell of the vehicle as shown in Figure 4-9 and would receive heated coolant fluid from the coldplate loop. As the fluid passes through the radiator, the heat emitted by the radiator surfaces is extracted from the coolant. The fluid then completes the loop by being resupplied to the coldplates at a lower temperature.

The performance characteristics of radiators are complex in that they are highly dependent on a number of specific thermal design and geometric parameters. Detailed design is not within the scope of this study; however, a conceptual analysis based on a typical design was undertaken to demonstrate the feasibility of radiators for this application. The results indicated that for the range of operation of the space tug astrionics, the required emitting surface area was 90 to 125 square feet. This is consistent with values of 25 to 50 btu/hr./ft.^{2} of surface area quoted in References J-2 and J-4 for similar applications and reveals that a major portion of the external surface of the astrionic module would be radiator surface area. As was treated in Section 4.1, the external skin surface was a primary design item of the passive thermal control concept. Thus, since the radiator now forms this surface, it must have optical properties compatible with both active and passive systems. The radiator demands a high emissivity and low α_S/ϵ ratio for effective operation. Initial estimates place the value of $\alpha_S/\epsilon < 0.4/0.9$ for the allowable surface area. This does

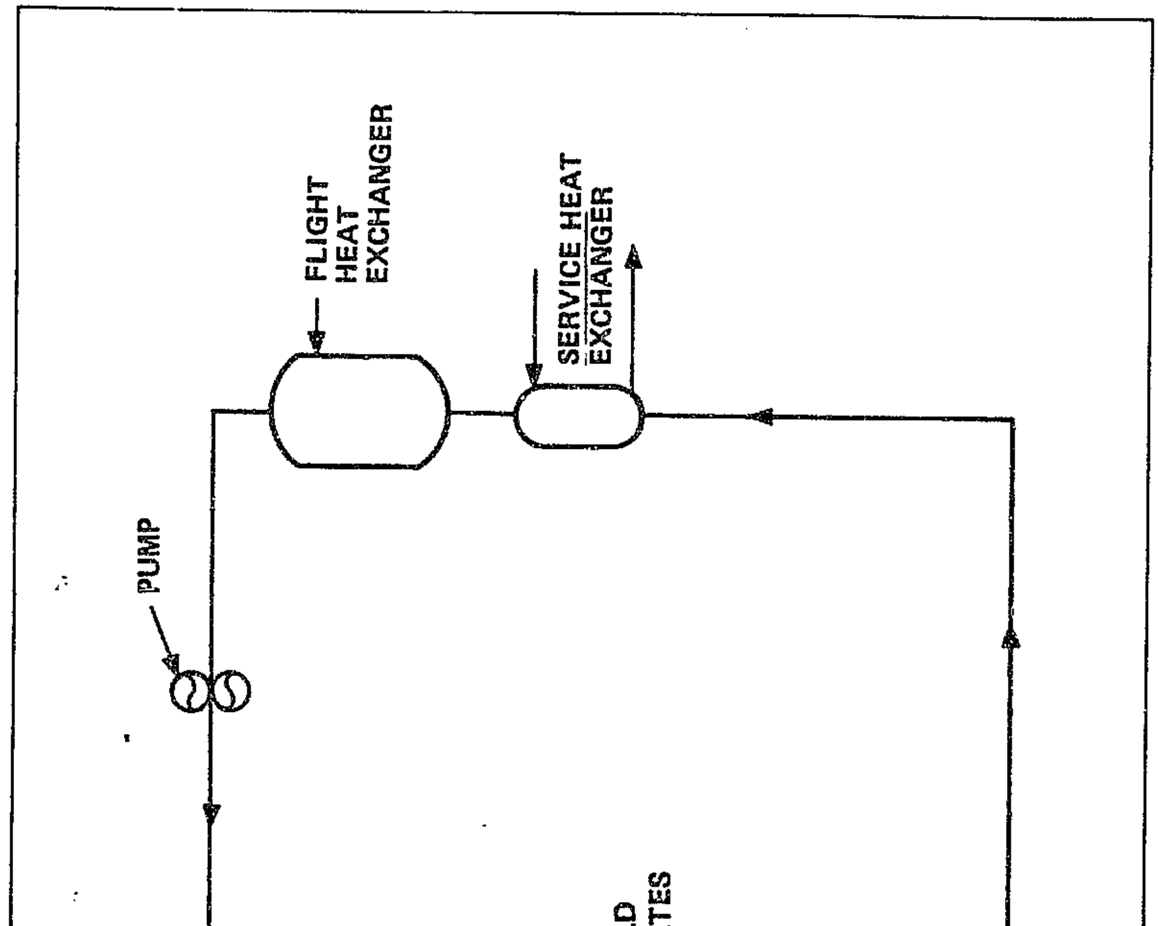
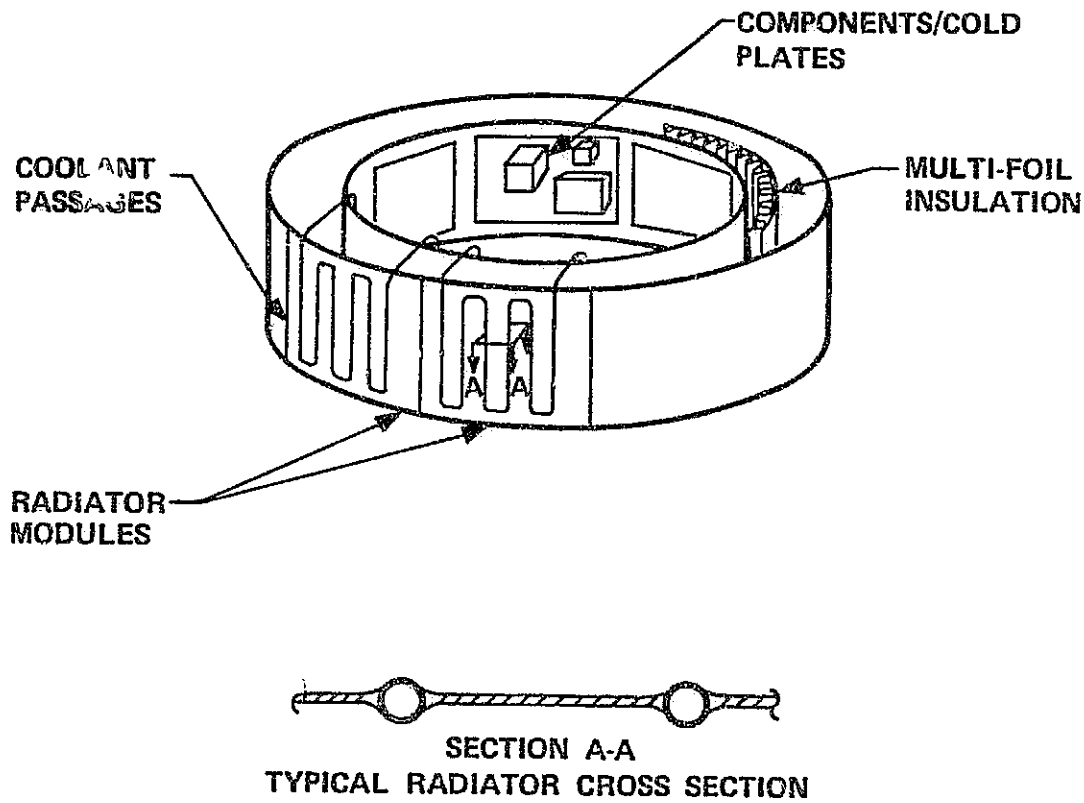


Figure 4-8. Typical Closed Loop Fluid Circulation System

not appear consistent with the $0.8 \leq \alpha_S/\epsilon \leq 1.2$ determined previously. Hence, use of a space radiator in this application would also require the use of louvers as discussed in Section 4.1. An additional benefit can be derived from the use of louvers if the range of actuation is made consistent with the desired temperature range of the active system radiator. The louvers become the control mechanism for the active system and additional fluid control techniques would not be required.



NOTE: 1. RADIATORS ARE AVAILABLE IN MODULAR FORM AND WITH VARIOUS COOLANT PASSAGE DESIGNS

2. COMPONENT ACCESS CONCEPTS ARE SHOWN IN APPENDIX I

965-138

Figure 4-9. Radiator/Outer Shell Integration

Estimated weights for radiators are in the range of 0.5 to 1.25 pounds per square foot (References J-2, J-3 and J-4) depending on the particular design. The top end of the range includes an allowance for coolant fluid within the radiator and a double wall thickness on the external side of the fluid passage. The double wall thickness is an added factor for protection against meteoroid penetration and contributes only some 5-10% of the cited weight. Louvers would contribute an additional 0.5 to 1.0 pounds per square foot (Reference J-5). Based on a maximum surface area of 125 square feet, using a weight factor of 1.25 and 0.75 pounds per square foot for the radiators and louvers, respectively, total weight for the radiator/louver heat sink concept would approximate 250 pounds.

The principal advantages of the radiator approach is that these exchange devices are not mission duration limited. Since no expendables are required, the device will function continuously with virtually no requirement to be refurbished or recharged between missions or mission phases. This is an almost overwhelming plus factor when considering the multi-mission reusable role of the space tug vehicle. The usual disadvantages of radiators such as 1) reduced performance due to possible degradation of the surface optical properties and 2) reduced reliability due to possible meteoroid penetration can be overcome by proper engineering design and maintenance philosophy. For example, a large amount of data is available on the degradation of optical properties in the space environment. The band of louver control can be selected to compensate for the predicted degradation. This will be simplified further by maintainability provisions that are required by space tug design groundrules.

4.2.2 Sublimator

A sublimator is a two fluid heat exchanger in which an expendable fluid, on being exposed to the space vacuum, freezes and then sublimes to vapor. The heat of vaporization is removed from the primary coolant fluid flowing in adjacent passages. The integration of the sublimator into a typical system is shown in Figure 4-10. Temperature control is generally obtained by modulating the amount of coolant fluid through or around the sublimator or, alternately, by controlling expendable fluid to the sublimator. The expendable fluid is supplied from a pressurized reservoir. The amount of fluid required is dependent on the total quantity of energy to be dissipated (over the duration of the specific mission) and the latent heat of vaporization of the secondary fluid itself. The relationship is:

$$m = \frac{Q}{h} \times \theta$$

where Q is the continuous (average) heat dissipated, btu/hr
h is the latent heat of vaporization, btu/lbm
θ is the active mission duration, hrs.

Of the candidate fluids, water offers the greatest heat dissipation per unit weight (1050 btu/lbm) and will therefore be used in determining mass requirements. As stated in Section 2.0, the electronic cooling requirement is for 1.3 kilowatts. In addition to this, it will be conservatively assumed that the maximum external environment net heat input to the system will be less than 0.75 kw in each case. Using the above relationship, the expendable water mass requirement was determined for each mission and compiled in Table 4-1 along with the weight of the supporting hardware peculiar to the sublimator system-storage tanks, pressure regulation system, control hardware, etc.

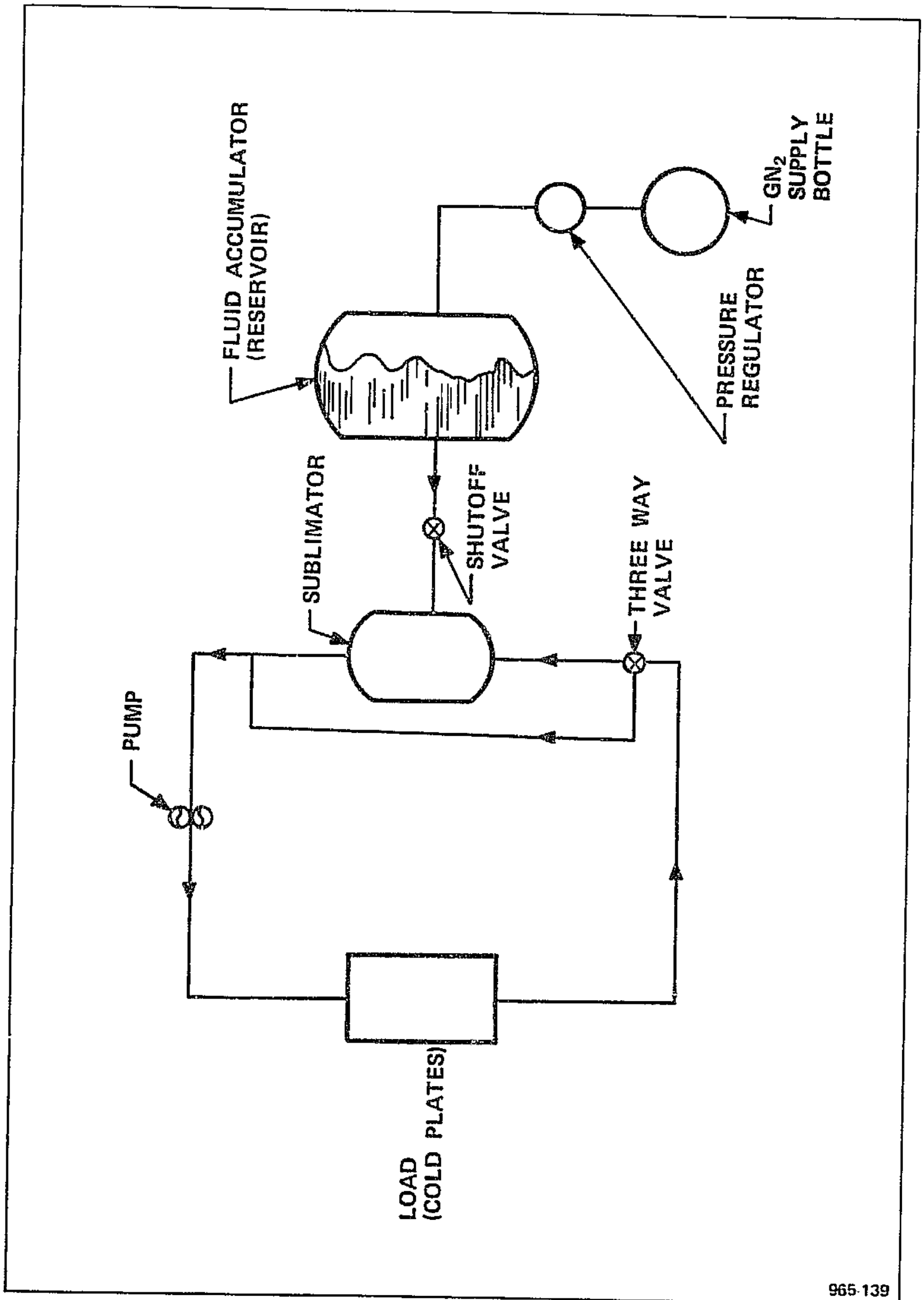


Figure 4-10. Sublimator System Schematic

The fuel cells of the electrical power system produce water as a by-product of the power conversion process. This fluid could be recovered and used as a part of the expendable water supply; however, it is at an elevated temperature and would require cooling itself before it could be used in the sublimation process. The additional water which would have to be stored to cool this by-product water plus the necessity of adding a mechanical recovery system would greatly offset any benefits gained by recovering and using the waste water in the heat exchanger subsystem.

4.2.3 Sublimator/Radiator Trade

The degree of thermal control obtained from each of these candidate heat exchanger systems is satisfactory for the astrionic module application. Hence, the trade to determine the optimum system must be made on the basis of weight, cost, reliability and applicability to intended mission. The weight of the radiator system was determined to be a maximum of 250 pounds, while that of the sublimator system varied from 107 to 530 pounds. Neither system appears to offer commanding advantage from a relative weight standpoint. Initial hardware costs would likely favor the sublimator system. In terms of reliability, a detailed design evaluation would be necessary to obtain explicit values, but in general terms it would essentially be the possibility of meteoroid puncture of the radiator coolant passages versus the reduced reliability of additional sublimator components (valves, pressure regulator). The overwhelming trade item to determine the ultimate choice is considered to be the applicability to the intended missions. The space tug concept is that of a utility vehicle to be used in both a support system and primary vehicle role. It must be capable of performing a spectrum of missions and be reusable with a minimum of refurbishment over an extended lifetime. The space radiator offers distinct advantages over the sublimator concept in that it requires no consumables and is therefore not limited by the duration of a specific mission and requires no refurbishment between missions. The sublimator system requires replacement of consumed fluids and must be serviced between (before) each new mission, the amount of fluid needed being dependent upon the particular mission requirements. Thus, the recommended heat exchanger device for further thermal control system study (see Section 4.4) is the space radiator. Table 4-1 A summarizes the heat rejection trade.

Table 4-1.

Mission	Duration, Active Component Operation (Hours)	Minimum Cooling Capability (Kilowatts)	Stored Water (Lbs)	Tankage at 20% of Stored Water (Lbs)	Sublimator (Lbs)	Pressure Regulation and Control Hardware (Lbs)	Total Weight (Lbs)
I Synchronous Orbit							
1) Expendable Mission (1 Tug)	19	2.05	135	27	25	25	212
2) Reusable Mission (2 Tugs)							
No 2 Tug	60	2.05	400	80	25	25	530
No 1 Tug	18	2.05	120	24	25	25	194
II. Orbital Operations	24	2.05	160	32	25	25	242
III. Planetary (2 Tugs)							
No 1 Tug (return)	24	2.05	160	32	25	25	242
No 2 Tug (dispose)	7	2.05	47	10	25	25	107
IV Fourth Stage	16	2.05	107	22	25	25	179

Table 4-1A. Active Heat Rejection Options

	ADVANTAGES	DISADVANTAGES
• WATER BOILER	• NO MICROMETEOROID EXPOSURE • NO ORIENTATION CONSTRAINT • COMPACT	• REQUIRES EXPENDABLES • CONTROL MORE DIFFICULT • RELATIVELY HEAVY
• SUBLIMATORS	• NO MICROMETEOROID EXPOSURE • NO ORIENTATION CONSTRAINT • COMPACT	• REQUIRES EXPENDABLES • REQUIRES ADDED SUPPORT EQUIPMENT
$\boxed{\text{• RADIATORS}}$	$\boxed{\text{• NO EXPENDABLES REQUIRED}}$	• MICROMETEOROID EXPOSURE

4.3 THERMAL CONTROL CONCEPT DESCRIPTION

The thermal control concept that has evolved to this point consists of a passive technique for maintaining acceptable component temperatures during power-down modes and active techniques to provide the additional capability needed during the active mission phases. This section attempts to integrate the two techniques into a total conceptual design for the astrionic module.

The recommended thermal control concept for the astrionic module is shown in Figures 4-1 and 4-2. The electronic components are mounted to coldplates which are fastened to the primary (load bearing) structure. When the components are powered up, a pump circulates the coolant fluid to space radiators which reject heat to the deep space environment. The radiator consists of coolant passages implanted in a thin metal shell which encircles the primary structural ring as shown in Figures 4-1 and 4-9. This outer shell forms a thermal radiation barrier to the external environment. Further isolation between the internal compartment and the external environment is afforded by low conductivity standoffs which fasten the outer shell to the inner ring and through the use of multifoil insulation between the two elements. It should be noted that the outer shell is not a structural member, its only purpose being to present a controlled surface to the external environment for thermal and for micrometeoroid protection.

A system of louver panels cover the radiator surface area and will be fastened directly to the segmented radiator panels. These individual modules are compatible with the accessibility options discussed in Appendix I. The louvers are driven by bi-metallic thermal actuators sensing temperature of the underlying surface. The segmented louvers function independently, giving a degree of local control around the vehicle. Individual louver/radiator panels may be removed for the expendable missions to provide an optimum combination of thermal control, weight and cost.

4.3.1 Pressurization

Liquid pressurization of the coolant loop can be provided by two methods:

- GN_2 pressurization through a bladder/liquid interface
- Mechanical pressurization using a spring loaded bellows type accumulator.

The latter requires no consumables and is therefore preferred in this application. Some capacity, though limited, is available within the bellows accumulator for providing makeup fluid in the event of small leaks.

4.3.2 Temperature Regulation

Coolant temperature control can be achieved through a number of schemes. The optimum approach appears to be the utilization of the louvers to regulate the amount of radiating area. The actuators would have an operating range consistent with the required radiating temperature of the emitting surface. An alternate method of control is to use a fluid by-pass around the heat exchanger when cooling is not required, identical to the approach previously illustrated in Figure 4-10 for the sublimator system.

4.3.3 Service Heat Exchanger

An auxiliary heat exchanger is included to allow conditioning of the onboard system by a space element when the tug is docked with the element and for ground conditioning during system checkout.

4.3.4 Coolant Fluid

The coolant fluid selected must have a low freezing (pour) point as well as the other desirable properties of low viscosity, high specific heat and compatibility with other materials.

4.3.5 Reliability

Principal reliability aspects generally common to liquid loop systems are the provision for assured containment and pressurization of the liquid volume and maintenance of circulation. The degree of reliability necessary or desirable and the resulting penalties for achieving this reliability is a study in itself and will not be considered here beyond the point of mentioning that because of its critical nature and reliability, the recommended system includes a redundant coolant pump.

4.3.6 Weight, Size

Rough estimates of weight and dimensional properties for the major system components are compiled in Table 4-2.

Table 4-2. Estimated Weight and Dimensional Properties of Thermal Control Components

Component	No.	Weight (Combined)	Size
Pump (150 watts)	2	50 Lbs	8" x 6" x 6"
Coldplate	8 (Max)	200 Lbs	48" x 36" x 1"
Accumulator	1	30 Lbs	10" x 12"
Radiator	8	147 Lbs	48" x 44" x 2"
Louvers	8	88 Lbs	48" x 44" x 3" max.
Service Heat Exchanger	1	10 Lbs	6" x 5" x 3"
Fluid		50 Lbs	
Multilayer Insulation		25 Lbs	
Plumbing, Misc. Hardware		60 Lbs	
Total		660 Lbs	

Values in this table are rough estimates for general planning purposes only. Reference Figure 4-2 for a schematic of this concept.

4.4 LUNAR MISSIONS

In previous paragraphs (Section 2.0) the space tug missions were categorized to include a number of common phases of thermal environment. It was stated that the lunar missions could not be included in this generalization because of their unique environment and would, because of time limitations, not be considered in the conceptual studies. This section presents a brief discussion of the lunar missions attempting to point out some of the aspects which must be considered in order to provide satisfactory thermal control.

The two prime factors in determining thermal control requirements are the electronic component thermal characteristics (power dissipation, allowable temperature, etc.) and the thermal environment to which the vehicle will be exposed. The former has been defined for the lunar landing mission and from a thermal standpoint is no more severe than for the other space tug missions. The external thermal environment however is both more severe and more difficult to define.

The lunar mission as currently defined includes both a lunar orbit and a surface operation. In each phase, there is an active period in which the astrionic components are powered up and an inactive period in which the entire vehicle is in a quiescent or storage mode.

- Lunar Orbit The determination of the orbital heating environment for a vehicle orbiting the moon is carried out in much the same manner as for the earth orbit cases. The orbit parameters defining the worst possible thermal environments are first determined, this information then being input to a digital computer program to define the exact heating values. Perhaps the most significant difference between the lunar and earth orbit cases is that, because of its relatively uniform atmosphere and short period of revolution, the effective emitting temperature of the planet earth is generally uniform. The moon, on the other hand, is for all practical purposes void of an atmosphere and experiences a relatively long day (approximately two earth weeks). This results in large variations in temperature (>500 F degrees) over the surface, causing large variations of lunar thermal heat flux with latitude and longitude which must be taken into consideration. The orbit parameters must be carefully selected in order to avoid significant errors in the thermal analysis.

- Lunar Surface Operation – Determination of the thermal environment for a vehicle on the lunar surface is also complex. The two week periods of sunlight and darkness present exactly opposite environmental extremes. During the lunar day, the high temperatures of the surface (up to 250°F) and large "view factor" result in values of lunar emission striking a surface vehicle in excess of the normally predominant solar incidence. Areas with irregular surface conditions (mountains, craters, etc.) may intensify the heating conditions. In the lunar night, surface temperatures approach -250°F.

As with the other space tug missions, the lunar missions include extensive periods in which the vehicle astrionics is in a quiescent or storage mode (up to 180 days in lunar orbit and 42 days on the lunar surface). Thus, prime emphasis should be placed on passive thermal control techniques. The ultimate design of a vehicle which must withstand both the extremes of the lunar surface environment as well as lunar orbital heating will necessarily include extensive thermal protection and conditioning schemes in excess of those required for the other space tug missions. This could be accomplished with the baseline concept as defined herein with additional conditioning techniques (e.g., sublimator system) added in kit or modular form to provide capability for meeting the additional requirements of the lunar mission.

4.5 FUTURE TECHNOLOGY

The thermal control system/concepts recommended in this study are present state-of-the-art. It is not anticipated that advancements in technology within the time frame of the space tug evaluation will significantly affect these techniques. Potential refinement and demonstration of practicality of known concepts, plus the possibility that new information gained in the evaluation of the space tug will allow consideration of different techniques, make it advisable to mention some "future technology."

The heat dissipation function in the space environment is essentially limited to two processes:

- Thermal radiation to deep space
- Utilization of phase change energy, or sensible heat capacity, of a stored, expendable fluid.

All cooling systems must eventually come to one or both of these two processes; hence, most "new techniques" are either refinements of old methods utilizing these processes or new ways to utilize them. Some of these techniques which could become applicable to the astrionic module are:

- Thermodynamic Cycles – Active conditioning systems are possible which use methods of removing heat from a source to a sink at a higher temperature. Systems employing gas and vapor cycles may be used in conjunction with space radiators to decrease the required emitting area of the radiator surface. In general, these systems require more complex equipment than other active conditioning systems and none have been demonstrated in an actual space vehicle.

- Thermoelectric Cooling – Similar to the thermodynamic cycle, systems employing the thermoelectric effect remove heat from a source to a sink at a higher temperature. Both of these techniques are perhaps more correctly called "heat pumps" in that they do not in themselves dissipate heat but effectively improve the thermal link between source and sink. These systems have a common undesirable element, a very high ratio of power input to heat pumping rate. Improvements in this condition depend on the development of new and better materials.

- Heat Pipe – The heat pipe is a highly effective fluid heat transfer device which is self-contained, has no moving parts, and can transport many times as much heat as a solid conductor of the same cross-section. Essentially, the heat pipe may be thought of as a very low resistance heat conductor. Potential use appears best on the electronic component level in improving heat rejection to a primary thermal conditioning system.

- Utilization of Hydrogen Boil-Off – A propulsion stage in space which contains residual liquid hydrogen receives thermal energy from solar and other sources, causing the hydrogen to evaporate. This boil-off could conceivably be used to supplement existing heat rejection systems or in some cases provide the sole heat dissipation process.

- Thermal Control Coatings with Variable Optical Properties – An advanced concept which has received considerable attention in recent years and which may become a candidate for replacing the louvers, on the recommended system, is the use of thermal control coatings with variable optical properties. The α/ϵ ratio of these materials changes reversibly to present a "controlled" value consistent with the desired temperature. Some of the mechanisms involved include polarization, change in transparency, chemical effects and electroluminescence.

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APPENDIX K

SPACE TUG CHECKOUT CONSIDERATIONS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

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1.0 INTRODUCTION

The purpose of this appendix is to present the results of the initial study of the space tug checkout function. The first step of the study was to determine the checkout requirements for the system. Based on the checkout requirements, various types of tests to be provided were defined. Then a preliminary analysis of checkout methods was performed to establish system configurations which fulfill the space tug requirements.

2.0 GUIDELINES AND CONSTRAINTS

The checkout system study was performed based on the following guidelines and constraints:

1. The tug is based and maintained in space and on the ground.

2. The tug may be configured for manned or unmanned missions. In the unmanned configuration, the tug will be operated automatically or by remote operation from the earth or other space element.

3. The tug astrionic module design shall minimize the need for ground support.

4. The tug shall be capable of going from the quiescent state to a fully operational state within two hours.

5. The astrionic module will be packaged to allow a remove and replace maintenance and reconfiguration concept in space.

6. The design goal of the checkout function will be to locate failures to a lowest replaceable unit (LRU) consistent with the philosophy of modular astrionics.

7. The astrionic system will be capable of checkout and monitoring for the total tug consistent with the tug maintenance concepts.

8. Between missions, maintenance will be performed as required to upgrade the reliability to the required level. During the mission, maintenance will be limited to switching to redundant paths.

9. The following definitions of failure criticalities establish a baseline for checkout requirements:

   - Criticality I failure is one that jeopardizes the safety of the crew.

   - Criticality II failure is one that causes a primary mission abort.

   - Criticality III failure is one that neither jeopardizes crew safety nor results in primary mission abort but causes other mission impact. (For example, a failure which causes loss of vehicle autonomy.)

3.0 SUMMARY OF RESULTS

Test requirements and test definitions for the space tug were examined. Two approaches to providing an onboard checkout capability for the tug are presented: 1) centralized checkout and 2) built-in test equipment. The advantages and disadvantages of each approach were appraised based on the overall tug requirements and the projected astrionic systems. It was concluded that the centralized checkout approach provides major advantages and should be considered the prime checkout tool. However, it was pointed out that built-in test equipment would be more advantageous in some areas. Therefore, the tug checkout function should be a combination of centralized checkout and built-in test equipment.

4.0 DETAILED ANALYSIS

4.1 CHECKOUT REQUIREMENTS

Checkout for the tug includes the process of determining the operational capability of the various tug systems, detecting and isolating malfunctions, and reverification of operational status after a maintenance sequence. Checkout will include pre-flight, pre-mission, and operational checkout, as required in the various mission phases.

The tug checkout function will be accomplished by onboard equipment and will provide the means for automatic and manual operation. The status of tug systems, as determined by the checkout function, will be provided to the crew and through the telemetry system to the ground or other space elements.

4.1.1 Test Requirements

The following requirements provide the basis for definition of checkout capabilities:

- Detection of a critical malfunction shall result in switching to an alternate unit, module, path, or method, in order to complete the mission or save the vehicle.
- The system will monitor various parameters for trends and determine a prediction of the time that an unsatisfactory condition will exist. The capability shall exist to alert the crew and mission control of equipment degradation or potential failures.
- A preset testing sequence utilizing test stimuli to a unit, either generated internally or externally to the unit, shall be provided. The testing sequence will include analysis of the unit outputs.
- Diagnostic testing using data from a number of sources shall be provided. Diagnostic testing will be required when the cause of an anomaly is not readily apparent from individual monitoring points.
- The design goal for the checkout function will be to isolate malfunctions to a lowest replaceable unit for any subsystem.
- Capability to test the entire vehicle shall be provided.

- The checkout function shall be capable of remote activation and monitoring.

- The checkout function will be designed so as to not adversely affect the component or subsystem under test.

- Onboard vehicle checkout equipment and test techniques shall be compatible with launch and support facilities, insofar as possible.

4.1.2 Test Definitions

There will be several general types of tests required to fulfill the checkout objectives for the tug systems. Definition of these tests will further aid in decision-making regarding the checkout function configuration. These tests are discussed in the following paragraphs.

4.1.2.1 Overall Systems Test

The overall systems test will provide a complete vehicle systems test with a maximum capability to determine readiness of the equipment to successfully complete its assigned mission. This test will be performed at system activation, whether on the ground or in space at the end of a quiescent mission phase. It shall also be available for system checkout after maintenance and/or refurbishment. The test will be designed for flexibility to allow checkout for various vehicle, payload, and mission equipment configurations.

This test will be stored in the mass storage device and will be called into the computer main memory when needed. This test will perform a complete end-to-end checkout of all systems and subsystems. All redundant paths will be tested when practicable, along with the switching circuit for the redundant paths. The checkout system itself will be tested by diagnostic self-test programs or built-in self-test equipment.

The test will provide for trend analysis as well as detection of failures. Trend analysis will be performed by comparing data with preset limits and/or previously stored system test data. Data compression techniques of time and/or magnitude compression will be used for analysis of data. Data obtained will be retained in mass or main storage for engineering evaluation purposes and use in later checkout sequences.

In the event of the detection of an anomaly, the system will notify the crew (if manned) and mission control (ground or space elements) through the telemetry system. Maximum data concerning the problem will be transmitted. The same data will be stored in computer storage. After all pertinent data is saved, diagnostic routines will be performed to isolate the cause of the problem. The decision to continue or abort the mission will be made based on factors such as mission and payload criticality, backup capabilities available, and availability of manpower and spare equipments to perform maintenance.

4.1.2.2 System Functional Tests

System functional tests will provide the checkout capabilities required during active mission phases. These tests will be performed continuously on an interleaved basis with system functional operations. The tests will be co-resident with operational programs in the central processor main memory. The frequency of the various tests will be dependent on such factors as subsystem criticality, mission phase, and previous indications of equipment condition.

This test will establish the necessary confidence in tug systems to continue into following mission phases. It will analyze operational data to the maximum extent possible with minimal possible use of special tests or routines. Subsystem outputs will be tested generally, with points internal to subsystems tested less often. Internal points may be examined when outputs indicate a possible problem. Reasonableness tests will be performed. Data will be compared with data stored from previous missions or test runs in order to detect deteriorating conditions. Approximately 10% of all system test points will be monitored by this test. Approximately 2% of available tests will be utilized for trend analysis.

In the event of a malfunction, automatic switching (software or hardware controlled) to alternate methods or equipment will take place. Mission control and the crew (if manned) will be notified of the problem. The onboard data management function will have a capability of deciding whether to abort the mission or continue. This capability may be overridden from mission control or the crew module on manned missions.

4.1.2.3 Diagnostic Tests

Diagnostic tests will normally be stored in the mass storage of the data management subsystem. There will be overall system diagnostic tests, as well as subsystem dedicated tests. The diagnostics may be called into operation automatically by other tests of the checkout system, or they may be operated on command from mission control or the crew module.

The goal of the diagnostic tests will be to isolate any malfunction to an LRU. LRUs will be defined by the following characteristics:

- Isolation capability of the checkout system
- Physical size of the unit
- Ease of replacement of the unit

Isolation of a malfunction to a particular LRU will result in one or more of the following actions. Information concerning the malfunctioning LRU will be stored in memory, telemetered to mission control, and communicated to the crew module, if the particular mission is manned. If a malfunction occurs in a quiescent mission phase and maintenance facilities and manpower are available, the equipment may be repaired and the system reverified. If a malfunction occurs during a mission, then decisions will be made as to whether to abort or attempt to complete the mission with limited capability.

4.2 CHECKOUT FUNCTION CONFIGURATION

A primary consideration in the process of establishing the tug checkout configuration is the requirement for autonomous onboard checkout for the various tug modules and systems. The system must be versatile and effective in meeting test requirements for the multitude of vehicle configurations, missions, and payloads

There are two generally accepted methods of providing an onboard checkout capability. One approach is to use a central computer for test control and data analysis. The second approach provides built-in-test equipment (BITE) in each subsystem. Each of these methods has advantages and disadvantages. A compromise approach should attempt to use the best points of both methods to provide the most effective system at the lowest cost possible.

4.2.1 Central Computer Checkout

The central computer approach to onboard checkout offers the immediately obvious advantage of utilizing the operational equipment and functional data paths of the tug data management function described in Appendix C. Subsystem data already available in the computer for operational purposes may be analyzed for checkout purposes with a minimum hardware penalty. The data bus and interface units provide the means of controlling, interrogating, and analyzing data from the various tug subsystems.

The computer provides the capability of more detailed control and analysis of test data than does BITE. In the event of an apparent malfunction, the computer may interrogate different data sources to isolate the cause of the problem. A computer controlled checkout system may be programmed to work around transient indications of failures, whereas a pure hardware system might cause switchout of equipment on a transient.

Inherent in the computer controlled checkout is the capability of working around malfunctions by automatically utilizing alternate paths. Also, the computer system provides testing of different hardware configurations by changing the checkout programs. In the case of manned missions, the computer allows manual operation of the checkout system.

A centrally controlled checkout system provides the advantage of preserving test data. In the tug system, this data will be stored in the mass or main storage device. Storage of test data allows long term evaluation of equipment performance. The stored data may be telemetered to mission control at convenient periods for flight evaluation.

There are penalties associated with use of computer for checkout. One of these is the effort required to develop the checkout programs. All equipment must be analyzed in detail to obtain specific test requirements from which detailed programming flow charts must be developed. Then the programs must be written and checked out.

The impact on the processing system must be considered. Additional main storage capacity will be needed for the checkout programs. The computer speed must be such that checkout can be handled along with the operational programs. An examination of requirements based on past experience, however, indicates that the proposed tug data management function would be able to handle the tug checkout requirements. The storage capability will range from 5 percent to 30 percent of the total main storage requirement, depending on the mission. More comprehensive checkout programs may be stored in mass storage for extensive checkout during maintenance.

4.2.2 Built-In Test Equipment

BITE is test equipment designed and built as an integral part of the prime equipment. BITE may be designed to provide checkout and malfunction isolation capability to almost any desired level, though there is obviously a practical limit to how much BITE should be added. BITE offers the advantage, in addition to utilization in a system configuration, of being available for use at any maintenance location.

Use of BITE provides test decisions at the subsystem or sometimes a lower level. Contrasted to the centralized checkout approach, BITE would impose minimum burden on the tug data management function. Only status indications for various pieces of equipment would be forwarded to the computer via the data bus. Raw parameter data would not be forwarded. It follows that the previously discussed advantages in the centralized approach of storage of test data would not exist with this approach.

Redundancy for reliability purposes is a prime consideration in tug astrionics design. BITE and reliability redundancy may be designed as a joint function to provide both malfunction isolation and switching capabilities. Stimuli generation for test purposes will be more effectively provided by BITE circuitry as opposed to imposing the burden of stimuli generation on the central processor.

A checkout system utilizing BITE involves several major penalties. Use of BITE adds weight and size to the prime equipment. The additional circuitry involved in BITE increases the probability of failures. BITE must be designed to minimize the likelihood of BITE failures impairing total tug capability.

4.2.3 Recommended Checkout Configuration

It is recommended that, as definition of the tug continues, the design of the checkout system be pursued to determine the most cost effective combination of centralized versus BITE checkout. Based on the preceding discussions of requirements and checkout methods, it is recommended that centralized checkout should be provided as the prime checkout tool. However, all tug systems must be analyzed for test purposes since there will be areas where BITE will provide the most effective test capability. Also, there may be subsystem test requirements which impose an extreme penalty on the data management function. In these cases, BITE will be required at the subsystem level. However, the data management function should provide the overall test control.

A fairly recent advance in checkout capability is inherent in use of the data bus and its associated interface units. The interface units for the various subsystems may be designed to perform certain checkout functions, such as stimuli generation and limit checking. This would lessen the burden of checkout on the central processing system. The development of interface units with a checkout capability is a future consideration for use in the tug, along with the centralized and built-in test equipment methods.

The preceding discussion was based on the use of the data bus and interface units to provide the electrical interface connections for the tug data management function. If the data bus is not used, then the checkout configuration would be modified. Use of a centralized input/output function would impose a greater burden on data management in monitoring all the test points that would be brought to a centralized point. Checkout control would still be centralized. However, the checkout burden would be shifted further towards BITE at the subsystem level in order to limit the quantity of test points to be interfaced with the central processor.

The centralized checkout aided by BITE will be used across the spectrum of space tug design missions. This scheme can be used generally for any space element possessing a central computing capability.

APPENDIX L

SPACE TUG MAINTAINABILITY CONSIDERATIONS

IBM No. 69-K44-0006H
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1.0 INTRODUCTION

Maintainability should be one of the primary considerations in any development program. It is extremely important that no design decisions are made which would compromise the basic capabilities for testing and repair of the tug astrionics. Therefore, the effort in this program has been to review preliminary designs and provide necessary inputs to system designers to ensure that the system design includes consideration of basic maintainability criteria.

The maintainability philosophy and concepts have been developed to establish checkout and maintenance requirements. These requirements provide a baseline for maintainability design. The following areas received primary emphasis during the preliminary maintainability analysis:

- Test requirements
- Checkout function configurations
- Definition of lowest replaceable units (LRUs)
- Ease of access for maintenance

2.0 GUIDELINES AND CONSTRAINTS

The following guidelines and constraints provide the background for this study:

- The tug is based and maintained in space or on the ground.
- The tug astrionic module design shall minimize the need for ground support.
- The tug shall be capable of going from the quiescent state to a fully operational state within two hours.
- The astrionic module will be packaged to allow a remove and replace maintenance and reconfiguration concept in space.
- Remove and replace maintenance will be performed between missions. During missions, maintenance will be limited to switching to redundant paths.
- A standard physical interface for astrionic equipment and a standard electrical interface for all electrical equipment will be provided to simplify interchange of components and data transfer.

3.0 SUMMARY OF RESULTS

The preliminary maintainability design effort for the tug astrionics has been concerned with insuring that due consideration be given to the basic checkout and maintenance concepts for the astrionic equipment.

Basic maintainability philosophy requires an onboard checkout function with the following test capabilities:

- Assurance of equipment performance prior to missions.
- Providing continued confidence in equipment performance during active mission phases on a non-interference basis with system operation.
- Positive and rapid detection of malfunctions with isolation to a lowest replaceable unit (LRU).

The checkout configuration concept defined to provide these capabilities consists of utilizing the central processing capabilities of the data management function to the maximum extent possible. The checkout concept also includes the use of special built-in test equipment in some functional areas.

Primary maintainability design characteristics of accessibility and replaceability have received special attention during the preliminary packaging and layout studies. All candidate structural configurations provide the desired capability of facilitating ease of removal and replacement of equipment. The replacement capability is at two levels: the LRU level and the component panel level. The panels contain groups of related components.

A list of candidate LRUs has been generated. This list is based on the presently known functional and mechanical characteristics of the various equipment. The list will change as tug development progresses; however, the present list provides guidelines for future activities in development of the checkout function.

4.0 DETAILED ANALYSIS

4.1 MAINTAINABILITY PHILOSOPHY

For the purposes of this study, maintainability is defined to include those functions and/or capabilities which allow:

- Assessment of vehicle operational status
- Assessment of the "health" of the astrionic system
- Ability to detect and isolate incipient and actual failure conditions
- Ability to "work-around" failed equipments and resume normal operation
- Ease in the replacement of failed equipments

Obviously maintainability considerations for the space tug must embrace earth as well as space-resident operations. For this study, however, only space resident operations have been evaluated. The reason is twofold: (1) earth based operations have not been defined, and (2) space resident operations impose more stringent maintainability requirements.

Autonomous operation is a basic design goal for space tug. From a maintainability standpoint this requirement dictates the need for an onboard capability which allows checkout of the vehicle astrionics and system and, depending on the results, causes appropriate action to be taken to pinpoint the cause of failure and switch in spare operational units. For unmanned missions, this capability must be automatic since human intervention is not possible. For manned missions, the capability may be fully automatic or a combination of automatic and manual features. With the automatic capability on manned missions, crew intervention is obviously not required. However, the information and the ability needed for crew intervention should be provided. A totally manual capability would not suffice since crew reaction times are not quick enough to detect and correct failures in critical mission functions.

The guidelines presented in Section 2.0 concerning removing and replacing failed units during and between missions dictates the need for two types of repair capability. During the mission, repair of a malfunctioning unit is not accomplished. Instead, once the failure is detected, automatic switching from the failed to an operational unit is accomplished. Between missions, repair of a failed unit is accomplished on a physical "remove and replace" basis. Therefore, packaging and installation concepts for astrionic hardware must facilitate ease of replacement with due consideration of EVA operations.

4.2 MAINTAINABILITY CONCEPTS

The above maintainability philosophy for the tug establishes the basic requirements for test capability and the level of repair for the various echelons of maintenance. These maintenance requirements provide guidelines for the maintainability design of the tug astrionic module. Two maintenance echelons for the tug have been considered. The first is maintenance to be performed prior to missions or in quiescent mission phases. The second echelon involves the maintenance actions during active mission phases.

4.2.1 Quiescent Mission Phases

Maintenance during quiescent mission phases basically involves ensuring that the various systems are at the performance level required for the next mission or mission phase. The onboard checkout function should provide the capability of fully testing all systems for end-to-end. A diagnostic capability to isolate malfunctions to a replaceable item should be provided.

The astrionic module design should allow two levels of replacement. One level of replacement should be the black box or lowest replaceable unit (LRU) level. LRU identification must take into account the following:

- Isolation capability of the checkout function
- Physical characteristics of the units
- Ease of replacement of the units

The second level of replacement should be at the component panel level. If components of a given subsystem are grouped in parallel, reconfigurations at the subsystem level would be possible.

4.2.2 Active-Mission Phases

Maintenance in the normally accepted sense of the word will not be performed during active mission phases. The checkout function should provide automatic testing of all systems on a continuous basis interleaved with normal functional operation. If a malfunction occurs, the repair action will consist of automatically switching to alternate methods or redundant equipment.

Detection of a malfunction must be followed by a decision process to determine whether or not to continue the mission. The decision will be based on several factors. The major factors are mission and payload criticality, backup capabilities available, and availability of manpower and spare equipments to perform maintenance.

4.3 TEST REQUIREMENTS

Test requirements have been developed to implement the maintainability concepts discussed in the preceding section. The requirements are used to aid in maintainability design of the equipment. In particular, decisions regarding the checkout function design are based on the test requirements. The test requirements are grouped in three categories. Overall systems test requirements are those necessitated by quiescent mission phase maintenance concepts. System functional test requirements are those imposed by active mission phase maintenance concepts. The third category of requirements is for diagnostic tests to be utilized when a malfunction has been detected by overall or functional tests.

4.3.1 Overall Systems Test Requirements

The overall systems tests should perform a complete test of the various tug systems. The test should be designed to provide maximum confidence that the tug equipment will successfully complete the planned mission. The test operation should be flexible enough to facilitate checkout of varying vehicle, payload, and mission equipment configurations.

All redundant paths, as well as the associated switching circuitry for the redundant paths, will be tested. A capability of validating the checkout equipment itself must be provided. End-to-end testing should be performed utilizing normal functional input signals or test generated stimuli.

The test should provide for the detection of deteriorating conditions as well as discrete failures. Several data analysis techniques may be used to detect equipment deterioration. Data may be compared with preset limits or previously stored system test data. Data compression techniques of time and magnitude compression may be used for analysis of changing parameters. Data obtained must be retained for engineering evaluation and use in later checkout sequences.

4.3.2 System Functional Test Requirements

Testing during active mission phases must be a continuous process with test operations interleaved with normal functional operations. Although testing is a continuous operation, all parameters will not require testing at the same rate. The frequency of tests will be dependent on such factors as subsystem criticality, nature of the tested parameter, mission phase, and previous indications of equipment status.

The test operation should analyze operational data to the maximum extent possible in order to minimize the impact of checkout requirements on tug operations. Subsystem outputs will generally be used as test points, with points internal to subsystems examined less often. Reasonableness tests should be performed on various operational parameters.

In general, the same data analysis techniques used in overall tests should be utilized to detect deteriorating conditions during operational phases. Data comparison techniques and comparison with preset limits would be utilized. Data may be compared with previously stored data from both functional and overall tests.

The detection of a malfunction will require automatic switching to backup capabilities of either alternate modes or redundant equipments. Data concerning the problem would be provided to the crew (if manned) and to mission control.

4.3.3 Diagnostic Test Requirements

Diagnostic tests shall be developed to take maximum possible advantage of equipment and methods existing for operational purposes. Operational inputs and outputs of equipment should be used for diagnostic analyses where possible. This capability will be supplemented in some areas by the generation of test stimuli. In equipments where redundancy is available, the redundancy switching circuitry should provide indications of malfunctioning equipment.

The diagnostic tests may be operated automatically in the event of a malfunction, or they may be operated under control of mission control or the crew (if manned). The design goal of the diagnostic capability will be malfunction isolation to an LRU. Data regarding malfunctions will be retained in the astrionic equipment, telemetered to mission control, and communicated to the crew module on manned missions. In quiescent mission phases, where maintenance facilities and manpower are available, the equipment would be repaired and the system reverified. If a malfunction occurs during a mission, a real time decision would be made as to whether to abort or to complete the mission with a degraded capability.

4.4 MAINTAINABILITY DESIGN

The initial maintainability effort has been concerned with providing those maintainability characteristics which fulfill the basic test and repair concepts for the tug astrionics. These basic maintainability design characteristics include:

- Rapid and positive detection of malfunctions
- Rapid isolation of malfunctions to a replaceable unit
- Ease of replacement of equipment

These characteristics have been emphasized in development of checkout configuration concepts, mechanical layout and packaging designs, and the selection of a tentative list of lowest replaceable units.

4.4.1 Checkout Configuration Concepts

Analysis of tug requirements has led to a tentative configuration for the checkout function. This configuration utilizes the planned data management function in a centralized checkout concept. The existence of central processor capability along with the data bus and associated interface units provides an efficient test system with minimum additional hardware penalty.

Built-in test equipment (BITE) will provide a necessary complement to centralize checkout in some areas. In some subsystems BITE may provide a more cost effective test capability than a strictly centralized approach. There may be subsystem test requirements which impose too extreme a penalty on the data management function (refer to Appendix K for a more detailed discussion of checkout).

4.4.2 Layout and Packaging

Major emphasis in layout and packaging design has been concerned with providing maintainability features of accessibility and replaceability (see Appendix I for a detailed discussion of structures and layout). The layout design features components mounted on component mounting panels. Minimum tooling is required to replace components on panels. The panels are attached to the main structure by captive quick release type fasteners. Mounting of functionally related components on a single panel facilitates system reconfiguration by replacement of panels.

Two primary structural concepts have been evaluated shell structure and open frame. The open frame concept allows access from either inside or outside of the module. Some variations of the shell structure provide the same access capability.

All connectors, electrical and mechanical, are quick-disconnect types. The connectors should be coded and keyed to prevent accidental misconnections. All components and panels should be clearly marked with the following information:

- Part number, part name, and weight
- Location identification
- Connection identifications
- Pressure requirements
- Warning labels, where required
- Orientation

4.4.3 Lowest Replaceable Units

In order to establish a goal for design of LRUs, a list of candidate LRU's for the various avionic equipments has been developed. This list is a preliminary analysis of such characteristics as functional modularity, isolation capabilities, and ease of replacement. Obviously this list will be refined as development progresses. The list is broken into subsystem areas.

4.4.3.1 Data Management Subsystem

The following list delineates the LRUs for the data management subsystem:

- Central Processor Unit
- Bus Control Unit
- Configuration Assignment Unit
- Magnetic Tape
- Auxiliary Monitoring Computer
- Main Memory (Note 1)
- Display Memory (Note 1)

Note 1: In case of the memories, if monolithic technology is used, a lower level of replacement will be possible. This lower level will be a basic operating memory module.

4.4.3.2 Navigation, Guidance, and Control Subsystem (NG&C)

In the NG&C subsystem, the LRU level for each functional component generally divides into two areas: (1) sensors and (2) electronics.

- Inertial Measuring Unit (IMU) – The strapdown hexad IMU contains twelve sensors. Each of these sensors may be a replaceable item. The electronics for each sensor may be packaged with the sensor and thus be part of the replacement item.
- Star Tracker – The sensor head and the electronics are separate replaceable items for each tracker.
- Horizon Sensor – The horizon sensor contains two sensor assemblies and an electronics package. Each of these three items is considered to be an LRU.
- Landmark Tracker – The Landmark Tracker contains two candidate LRUs: (1) the gimbal assembly containing the sensor head and (2) the electronics. In addition, the sensor head itself may be replaceable.
- Laser Radar – The beam generator assembly and the electronics package are the replaceable items.
- Landing Radar – An antenna assembly and an electronics package are the replaceable items.

4.4.3.3 Electrical Power Subsystem

Each of the following components is considered an LRU for the power subsystem:

- Fuel cell
- Hydrogen tank
- Oxygen tank
- Battery
- D.C. regulator
- Battery charger

4.4.3.4 Electrical Networks Subsystem

The following are LRUs for the electrical networks subsystem:

- Standard interface unit
- Monitoring and auxiliary monitoring units
- Power distributor
- Auxiliary power distributor
- Junction box

4.4.3.5 Thermal Conditioning Subsystem

The following components of the thermal conditioning subsystems are LRUs:

- Coolant pump
- Service heat exchanger
- Coolant accumulator
- Component mounting panel
- Louver panel
- Radiator section

4.4.3.6 Command and Control Subsystem

The command and control subsystem breaks down into several areas as far as LRUs are concerned. Each passive component, such as antennas, diplexers, power dividers, etc., is a replaceable item. The electronics fall into three categories: (1) USB equipment, (2) VHF transceiver, and (3) other special equipments.

The USB equipment separates into four LRUs:

- Power amplifier
- Transponder module
- Modulator/demodulator
- Transceiver module

The VHF transceiver separates into three LRUs:

- Receiver module
- Transmitter module
- Ranging module

The third category of equipment consists of the following LRUs:

- VHF command receiver
- Command decoder electronics
- TV camera
- USB hi-gain antenna control
- Audio equipment (intercom)

APPENDIX M

SPACE TUG RELIABILITY CONSIDERATIONS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 269

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1.0 INTRODUCTION

This appendix presents the results of a study performed to:

- Describe reliability enhancement techniques which could be implemented into the space tug astrionic systems.
- Identify a reliability goal for space tug astrionics.
- Evaluate the reliability of space tug astrionic configurations which utilize basic reliability enhancement techniques.

2.0 STUDY GUIDELINES AND GROUNDRULES

Overall guidelines, groundrules and assumptions for the space tug study are presented in previous appendices of this report. Details or additions which particularly influence the reliability study are detailed in this section.

The definition of required astrionic module reliability must be preceded by the definition of total program goals. For this study, it was assumed that the astrionic module should have a 90% probability of causing no mission failures in 100 missions. Since $(\text{goal})^{100} = 0.9$, the required astrionic module reliability goal equals 0.999.

The extent and frequency of maintenance to be performed on the astrionic system will have a significant impact on system reliability requirements. By groundrule, the system must be capable of being maintained while in space residence. For this study it was assumed that the maintenance to be performed between missions would result in a completely operational system. That is, all system elements, both prime and back-up, are operational and all system elements are and will be operating within the range of their expected lifetimes. With this assumption, system eliability will be reset to the required level at the initiation of each mission. The limited maintenance during the mission (i.e., automatic redundancy management) enhances the probability of mission success but does not reset system reliability since all system elements are no longer functioning.

3.0 SUMMARY OF RESULTS

Of the four configurations evaluated, only one, the reusable synchronous orbit configuration, meets the assumed requirements. Through the use of higher sparing, all except the lunar landing configuration can meet the reliability goal.

A significant reliability problem exists with the lunar landing configuration. Implementation of full 1 + 2 sparing with 0.99 coverage does not enhance system reliability to an acceptable goal because of the long (~ 44 days) mission time. Relief of this problem can be realized in one of two ways: (1) either change the groundrule which precludes maintenance activity during the mission to reset reliability to an acceptable level, or (2) implement more sophisticated enhancement techniques such as increasing the amount of coverage. The optimum solution cannot be determined without a detailed identification and evaluation of resultant penalties associated with each solution.

4.0 DETAILED ANALYSIS

Redundancy is abundance, that is, having more than the minimum capacity to do the job. As used in the context of this study, it is the act of including more units in a system than the minimum required to do the job. System redundancy is mandatory for one or both of two reasons: (1) to achieve a higher probability of success for a given application than could be achieved with a bare minimum (simplex) system, and/or (2) to conform to specifications which dictate tolerance to one or more failures.

Two basic types of redundancy are recognized: masking and sparing. Masking is characterized by performance of the given function in three or more units and assumes the correct function will be represented by the majority in the event of any disagreement. The second general type of redundancy is sparing. It is characterized by performance of the function by a "prime" unit and includes in the system a means for determining if the "prime" unit is operating correctly and a means for replacing the "prime" unit with a "good" unit in the event of its malfunction.

For completeness the "theoretical" curves of Figure 4-1 are included to indicate some basic virtues and shortcomings of four commonly addressed hardware redundancy schemes.

One notices:

- The simplex (minimum hardware) system falls off most rapidly initially, gradually rounding its knee to asymptotically approach 0 at infinite time.
- All the redundant systems hold up (i.e., the curves are fairly level) before falling off.
- When the redundant system curves round their first knee, they fall off rapidly.
- That if one waits long enough the TMR system becomes eventually less reliable than the simplex system.
- That there is negligible difference between any of the systems if one waits long enough.

The preceding curves and observations are enlightening to the extent they provide upper bounds. However, in practical applications:

- It is academic what happens at times which are in excess of the total mission time – the important thing is how it performs within the mission time.
- Any redundancy scheme requires additional hardware for implementation which tends to minimize its advantages some implementations are more affected than others.
- The nature of the equipment (i.e., digital, analog and type of digital, analog) dictates the type of redundancy.

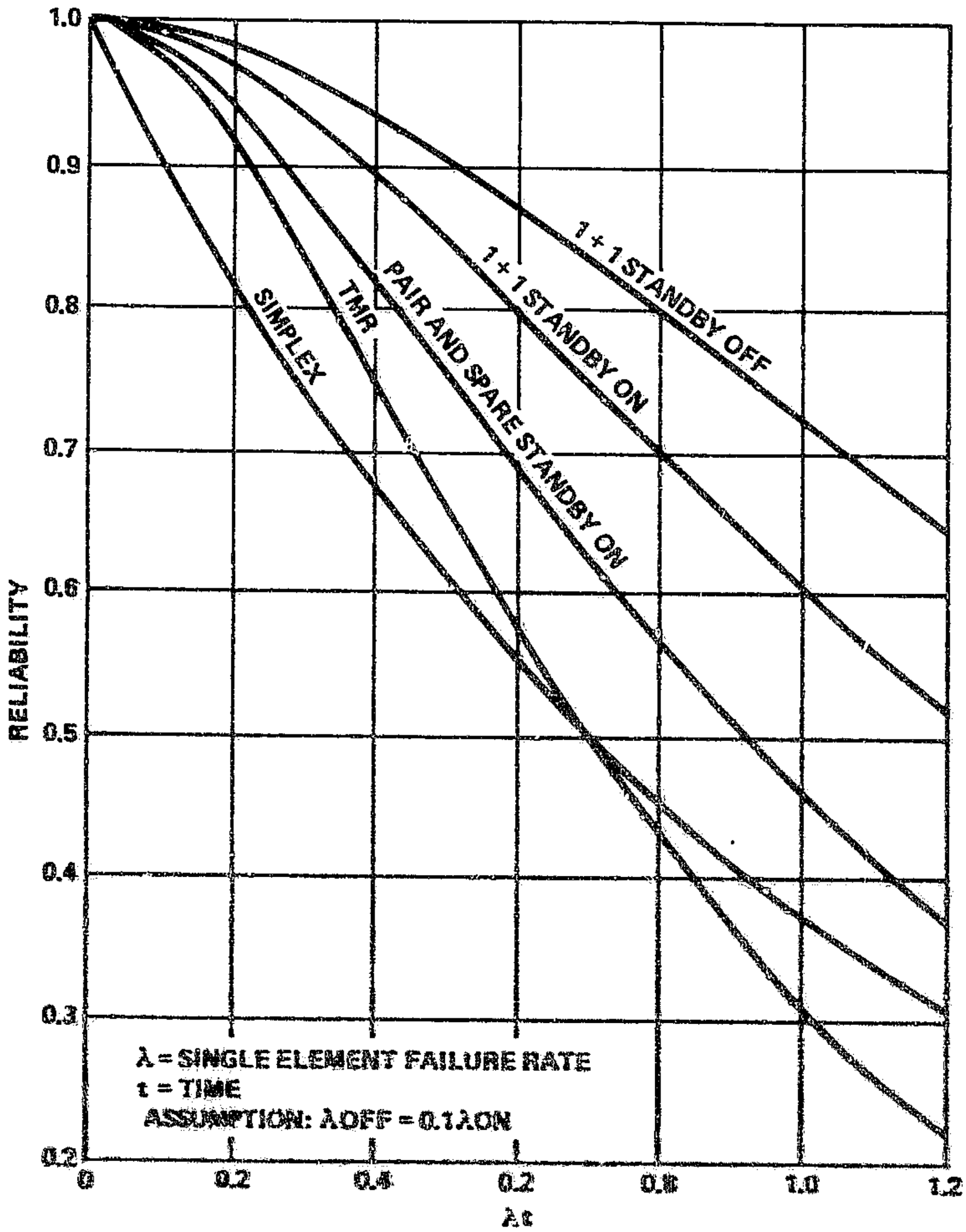

Figure 4-1 Typical Hardware Redundancy Schemes

Normally one is interested only in those portions of the curves which are in excess of 0.9 reliability (1 chance in 10 of failure). Assume for a moment that the vehicle system is composed of 10 major equally complex subsystems and each subsystem is composed of 10 equally complex units (black boxes). If the vehicle has a reliability goal of 0.9 then each of the 10 subsystems should have a goal of 0.99, roughly, and each black box a goal of 0.999. Most commonly the redundancy is applied at the black box level; consequently, the most common region of interest and applicability of the curves is in the region between reliability levels of 0.99 and 1.0. When attributes of various redundancy schemes in this region of interest are considered, and the additional detecting/switching hardware are included which increase the simplex failure rates, as in Figure 4-2, there are significant changes in the relative "goodness" of the various schemes. In Figure 4-2, the sparing schemes, 1 + 1 and 1 + 2, indicate a prime operating unit with one and two spares respectively. The numbers following in parentheses indicate the coverage of the system. Coverage is the probability of correctly detecting the need to switch, successfully switching, and effectively recovering from the malfunction. Considering Figure 4-2, it is apparent that:

- Sparing schemes offer the greatest potential for longevity only when the coverage is high.
- The graphs of the various schemes show that the scheme which is "best" for one reliability goal/mission/time combination may not be best for another combination.
- The nature of the hardware and the time interval the system can tolerate before malfunction recovery frequently preclude schemes that might otherwise be most desirable.

4.1 RELIABILITY ENHANCEMENT TECHNIQUES

4.1.1 Masking

The best known and widely implemented variation of masking is triple modular redundancy (TMR). It has the highly desirable virtues of essentially no delay in system operation when a malfunction occurs and of handling intermittent and hard failures with equal facility. On the other hand, TMR requires two units to function correctly at all times limiting its "endurance", and it requires considerable hardware to implement in some systems such as parallel organized digital processors.

Variations of the basic TMR scheme are feasible. For example, it is straight-forward to detect the failing module of the TMR triad and replace it with a fourth spare module.

4.1.2 Sparing

Duplication of hardware has been the most common type of redundancy scheme. It utilizes two identical units simultaneously performing the same function and compares their outputs as in the pair and spare curves of Figures 4-1 and 4-2. In the event of disagreement, both units are switched out (if there are not ancillary means of determining which has malfunctioned) and another pair or a simplex unit is switched in to continue the function. This type scheme has the advantage of being efficient from a design standpoint, since it effectively handles all types of malfunctions (except simultaneous occurrence of the same malfunction in two units) so that the coverage is essentially limited only by limitations of the compare switching recovery hardware. On the other hand, this type system is somewhat inefficient since properly performing hardware is switched out along with the malfunctioned unit.

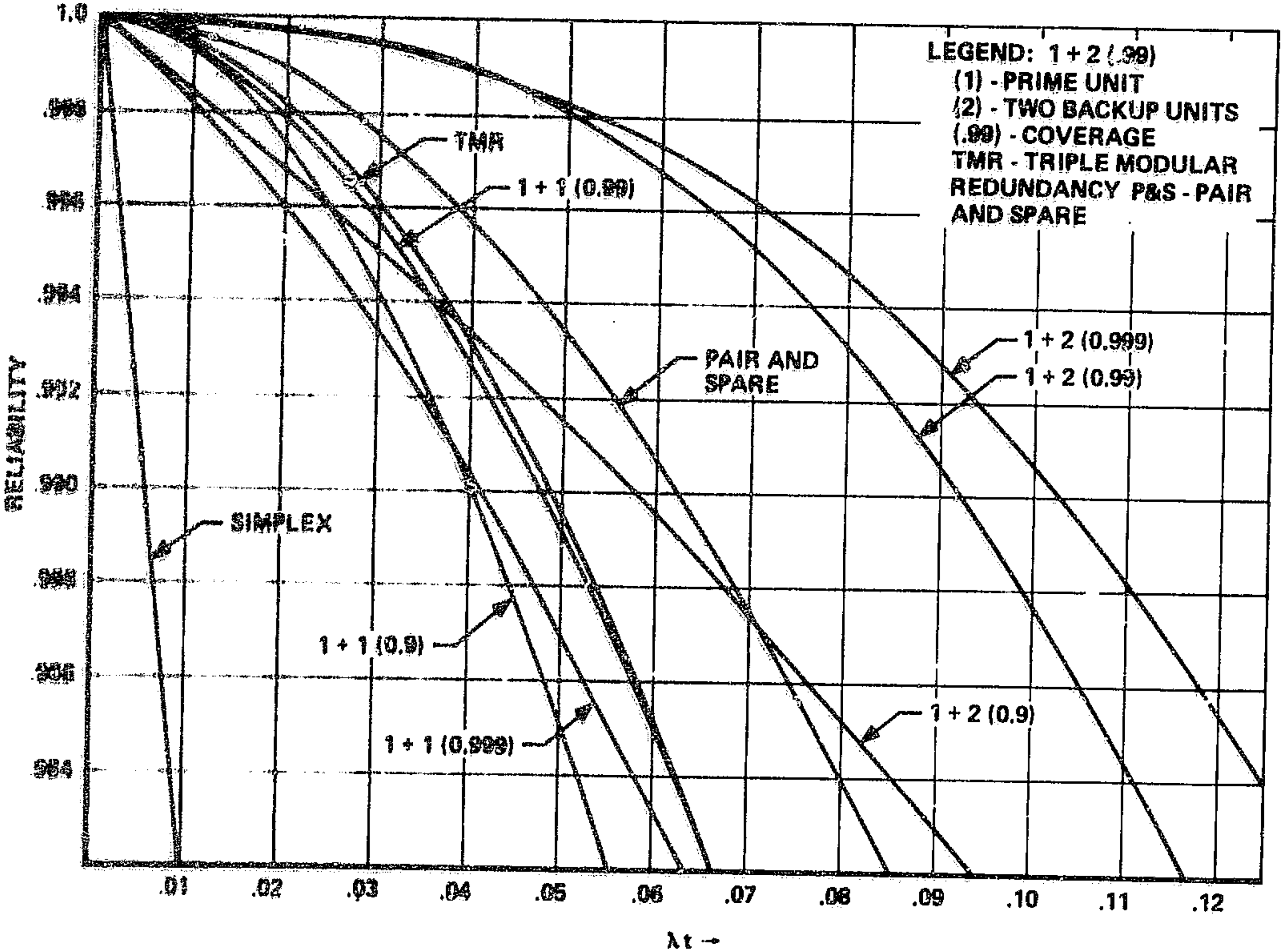

Figure 4-2. Example Reliability Redundancy Schemes

This type system has a variety of implementations all with one overwhelming requirement for success in application – namely, the "coverage" must be very near unity. Performance of the system is very sensitive to the coverage parameter. For example, predicted mission time at a fixed high reliability level may be degraded by a factor of 10 if the coverage is reduced from 99.6% to 90%. It is probably reasonable to say that sparing systems should not be considered for practical applications unless the coverage can be made to exceed 90%.

4.1.3 Software Coverage

In systems containing programmable logic, malfunction detection can be implemented by techniques such as limit tests, generation of test patterns, arithmetic operation checks, etc. This type coverage has the advantage of being relatively efficient from a hardware implementation standpoint. However, it requires time to perform and in depth knowledge of the system manifestations for all possible failures, and it is extremely difficult to analyze (in a quantitative sense) the amount of coverage obtained. Often it has the drawback of not being dynamic, i.e., not handling intermittent anomalies well. But this can be a very valuable technique to augment the failure detection hardware upon which the coverage techniques largely depend.

4.1.4 Dynamic Hardware Coverage

Dynamic hardware coverage has recently emerged from the research era and represents the best choice of hardware implementation. It is characterized by specialized design techniques such as dynamically checked logic (including the checkers), error encoding which enables dynamic detection and correction, multiple path reconfiguration, etc. These techniques are not amenable to retrofit but must be designed-in from the beginning. Additionally, quantitative knowledge of the technology failure modes and their probabilities of occurrence, the mission application parameters – duration, reliability level and recovery time, as well as the normal functional requirements must be known or defined and considered in detail. These designs require higher design costs but can result in significant weight and power savings as well as significantly greater longevity.

4.1.5 System Balancing

The foregoing has addressed the redundancy aspects of units irrespective of size. However, astrionic systems are composed of many different types of units and because of this diversity of the characteristics and functions of the elements comprising the system they are addressed individually in the system design. Thus reliable systems are frequently composed of elements which may have one type and/or level of redundancy and others another type. The reason for this is to obtain a balanced system design – that is, one which is free of having some elements much more likely to cause system failure than others. Also the characteristics of different devices make them more amenable to specific types of redundancy.

4.2 SPACE TUG ASTRIONIC SYSTEM RELIABILITY DESIGN

The system design effort for reliability on the space tug has been devoted primarily to selecting the appropriate reliability enhancement techniques for each of the subsystems, estimating the major systems parameters which determine the reliability, formulating appropriate mathematical models for several system configurations and evaluating their reliability effectiveness to arrive at an appropriately balanced system recommendation. The system which has evolved is primarily characterized by standby sparing type of redundancy for most elements with two notable exceptions: (1) in the area of the system memory; and (2) in the configuration of the inertial measurement unit. The design tool used for the quantitative modeling effort has been a time-shared computer terminal system (Reference M-1) which facilitates the modeling of a hypothetical system, the determination of system effects on reliability of the assumed parameters, and the modification of the parameters until a suitable design configuration is obtained.

The system parameters for the study are mission timelines, type of redundancy, power-on and power-off element failure rates, coverage (probability of detecting malfunctions, precluding their effect, and successfully recovering), and additional equipment to implement the redundancy. Of particular importance are the coverage and the additional equipment to implement the redundancy. These are assumed to be 0.99 and 50% of the unit being covered, respectively. Also, failure rates with power-off amount to one-fifth the rate with power-on.

4.2.1 Data Management Subsystem

In the areas of the system which are primarily digital, that is the computation/communication group consisting of the memory, central processing unit (CPU), configuration assignment unit (CAU), bus control unit (BCU), and the standard interface units (SIU), it is of critical importance that intermittent and hard failures be distinguished. Otherwise, the redundant elements may be ineffective and indeed in some instances detract from system operation. Also of critical importance is the ability to detect malfunctions as they occur during normal system operation. This prevents the propagation of errors and allows appropriate corrective action to be taken in order that recovery from the malfunction may be effected. To this end, the digital portions are to be designed with dynamically self-testing circuitry (Reference M-2). All data transfers between units shall be encoded in order that error detection and correction may be effected on these transfers. The configuration assignment unit will provide the redundancy management for the devices communicating with the data bus and will be largely automatic with backup provided by reasonableness tests.

4.2.1.1 Memory Organization

A relatively new concept for digital memory organization provides a very significant increase in error-free operating lifetime with a minimal increase in added hardware, together with excellent intermittent versus hard failure handling capability. The memory consists of a number of BOM (Basic Operating Memory) elements equal to the number of bits in the system word, plus some spare BOMs which may be switched in to replace defective BOMs. All stored words contain redundant bits such that all single bit errors may be detected and corrected during normal system operation. Thus, even though failed storage elements may

be randomly located on various BOMs, there is a high tolerance to this type failure, because the likelihood of a given word containing more than one of these bad bits is very low. On the other hand, assume that a catastrophic failure of one of the operating BOMs occurs, which effectively causes every bit which is read to be in error; e.g., a sensing circuit failure. The bits are corrected before the word is outputed to the system. Concurrently, the configuration assignment unit notes the multiplicity of bit errors from this BOM and institutes a recovery strategy to replace it. The replacement BOM may be loaded with good bits by utilization of the remaining word bits and the single-error correcting double-error detecting circuitry. Thus, the memory may be recovered intact from this catastrophic failure.

4.2.1.2 Central Processor Organization

The data management subsystem is composed of the central processor unit (CPU), the bus control unit (BCU), the configuration assignment unit (CAU), and the bit/BOM memory with its single-error correct/double-error detect circuitry. Each of these units will be designed to be dynamically self-checking; thus, all single failures in both the functional and the checking circuitry will be indicated immediately as they occur and before errors can be propagated to contaminate previously generated good data. Malfunction indication signals from each of the units will be received by the CAU. Normally, the first malfunction from a given unit will be assumed to be intermittent, and a re-try procedure will be initiated. Failure will be presumed when a sufficient number of re-tries does not result in successful performance of the operation. A recovery strategy will then be initiated by the CAU to transfer the data and status information to a standby and reconfigure the system to operate with the standby.

4.2.2 Inertial Measuring Unit

The hexad inertial measurement unit was originally proposed and analyzed by Gilmore (Reference M-3) in a master's degree thesis at MIT. Briefly, it utilizes six gyros mounted so as to have their input axes perpendicular to the faces of a regular dodecahedron. The axes of the dodecahedron make equal angles with the orthogonal triad vehicle axes. Thus, none of the six gyros' input axes are colinear with the vehicle coordinate axis, but each measures a known component of each vehicle axis motion. By virtue of this known geometric relationship, Gilmore develops relationships relating the output measurements of the gyros which may be used to verify their proper functioning, or alternatively, identifying any single, or any pair, of hexad element failures. Thus, the accelerometer and the rate gyro subunits would each consist of six accelerometers and associated electronics with system success obtained when any four are functional. Successful system function can indeed be obtained with only three of the six correctly functioning; however, identification of the third failed element cannot be obtained solely by comparison of their outputs. Auxiliary methods such as wheel speed sensing or reasonableness tests would need to be used.

4.2.3 System Encoding and Decoding

Extensive use should be made of error detecting/correcting encoding in order to maximize the system reliability with a minimum of hardware. For example, the bit/BOM memory organization is of little use unless augmented by circuitry which can correct and detect individual bit failures. The single-error correct/double-error detect (SEC/DED) memory input/output (I/O) section provides this function. The basic machine data word length is 32 bits. This word length requires 7 (redundant) bits be added in order that any single bit failure may be corrected and any double bit failure may be detected (but not corrected). Thus, each word stored in memory would be 39 bits. It is fortunate that the same physical circuitry which is used to encode words entering memory may be used to decode, correct, and detect bit failures of words leaving memory. Additionally, this circuitry can be designed to be dynamically self-testing in order to avoid its failing and erroneously "correcting" good data or failing to detect bad data. The data bus is another portion of the system where error encoding techniques will be utilized. Although the bus hardware may be extremely reliable, experience has shown that transfer of information between units in any system is susceptible to noise. This susceptibility will be effectively precluded by appropriate coding implementations for the bus.

4.3 SYSTEM RELIABILITY ESTIMATES

The following tug missions and their respective astrionic systems were evaluated and reliability estimates were determined:

- Expendable Synchronous Orbit
- Reusable Synchronous Orbit
- Lunar Landing
- Four Stage Saturn V

Table 4-1 and Figure 4-3 present the results of this evaluation.

4.3.1 Expendable Synchronous Orbit Mission

The "as defined" system configuration, which is virtually a simplex system has more than eight times the goal failure probability; however, this deficiency may be overcome by selective sparing of some of the larger failure rate components. Therefore, the fact that the configuration does not meet the goal is not deemed a significant design deficiency at this phase.

4.3.2 Reusable Synchronous Orbit Mission

The "as defined" system configuration for the tug I system configuration exceeds the goal. The tug II system configuration has problems which may be solved by sufficiently high level sparing or more desirably by increasing the coverage to make greatest use of the sparing levels presently defined with, perhaps, increasing the sparing in selected areas. Merely providing two spares for all of the critical units with the present coverage will not likely be sufficient to meet the reliability goal.

Table 4-1. Reliability of Configured Systems

Mission	Estimated Reliability (See Note 1)	Corresponding Unreliability x 10^3 (See Note 2)
Expendable Sync Orbit	0.991655	8.345
Reusable Sync Orbit		
Tug I	0.99139	0.861
Tug II	0.997231	2.769
Lunar Landing	0.974007	25.993
Fourth Stage	0.983597	16.403
Goal	0.999	1.000

Notes:

(1) Estimated reliabilities are for the astrionic system configuration as defined. The components for the astrionic system configurations are defined in the main body of this report (Section 4.0 – System Description).

(2) Unreliability = 1.0 minus the reliability.

4.3.3 Lunar Landing Mission

The "as defined" system has nearly 26 times the goal failure probability. This is due primarily to the combination of non-redundant critical elements and to the length of the stay (42 days) on the lunar surface. A significant increase in the probability of success of this mission can be gained by making all critical elements redundant and/or modifying the groundrules to allow refurbishment on the lunar surface before return liftoff. More than 90 percent of the mission unreliability is contributed by the long lunar stay. Modification of the groundrule would enable minor, straight-forward modifications to the "as defined" system to meet the goal. The probability of having all crew safety equipment (including redundancy) and all remaining functions operational at the conclusion of the lunar stay is greater than 0.96. Mission rules permitting, four missions out of 100 would require refurbishment on the lunar surface.

In the event it is not feasible to change the groundrule, replication of all critical system elements must be done and the coverage on virtually all elements must be increased, which entails additional hardware. However, this hardware increase penalty might be wholly or partially offset by enabling the sparing level required for some of the other mission configurations to be reduced. Thus, the penalty will depend, to a large extent, on the number and type of missions to be flown.

4.3.4 Four Stage Saturn V

The "as defined" system has 16 times the goal failure probability. This is due to the lack of redundancy in several units of the "as defined" system. Duplexing some or all of these will bring the overall system well within the goal.

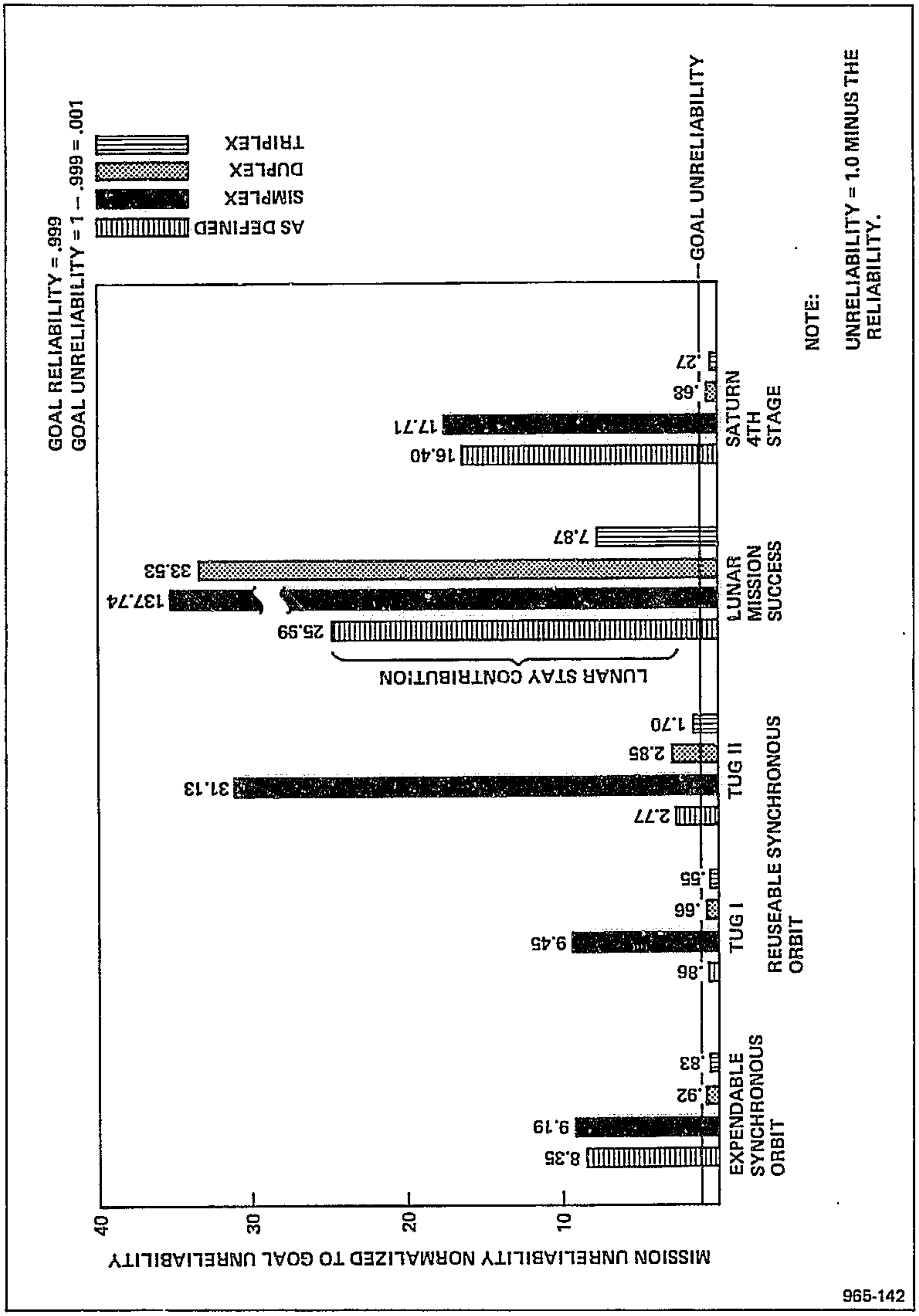

Figure 4-3. Astrionic Module Unreliability

4.4 SUMMARY AND CONCLUSIONS

Of the four configurations evaluated, only one, the reusable synchronous orbit configuration, meets the assumed requirements. Through the use of higher level sparing, all except the lunar landing configuration can meet the reliability goal.

Figure 4-4 shows the astrionic module reliability for the systems previously discussed. Estimates are also included of totally duplex and totally triplex configurations to indicate the reasonable limits to which the current de[illegible] may be extended. The pacing mission is obviously the lunar landing mission.

A significant reliability problem exists with the lunar landing configuration. Implementation of full 1 + 2 sparing with 0.99 coverage does not enhance system reliability to an acceptable goal because of the long (~44 days) mission time. A solution to this problem can be realized in one of two ways: (1) Either change the groundrule which precludes the maintenance activity during the mission to reset reliability to an acceptable level, or (2) implement more sophisticated enhancement techniques, such as increasing the amount of coverage. The optimum solution cannot be determined without a detailed identification and evaluation of resultant penalties associated with each solution.

The rationale for the 0.999 reliability goal is based on a probability of success for a 100-mission program which is at least 90 percent likely to have no mission failures. Thus, $\text{Goal}^{100} = 0.9$, Goal = 0.999. If the likelihood of no mission failures during the program (the 0.9) and/or the total number of missions in the program should be less and/or one or more mission failures could be tolerated, the goal may be correspondingly reduced. Figure 4-5 is a plot of the probability of no failures vs. number of missions for each of the "as defined" systems. As the program becomes more precisely defined regarding the number of missions of each type, these plots may be used to assess the total program success impact of each type of mission. For purposes of comparison, the Saturn Program with a goal of 0.992 for the astrionic system is included. The solid portion of this curve represents the probability of no failures for the current IU program (27 units). The dotted portion shows the effect on the probability of a 100 unit IU program.

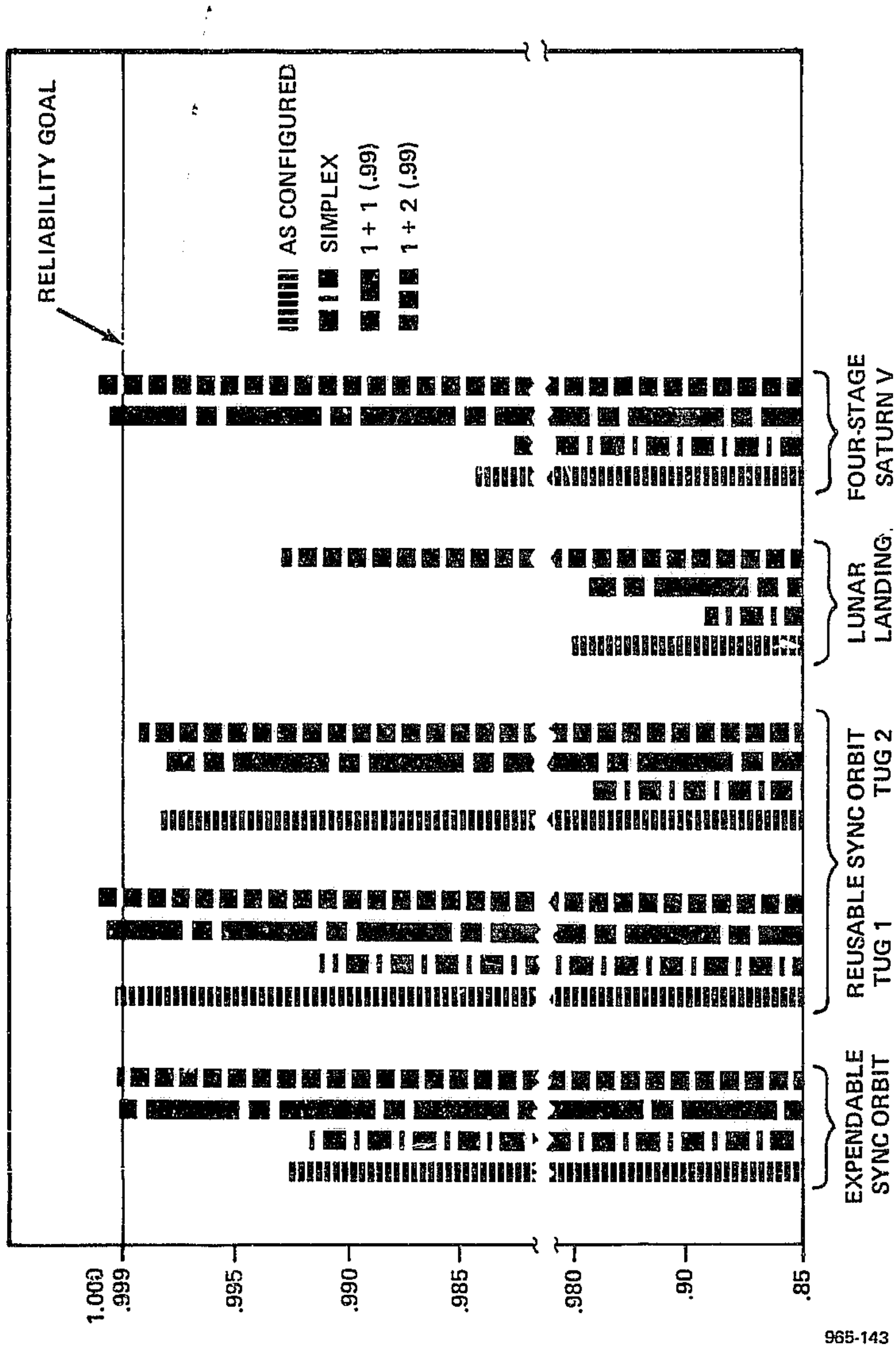

Figure 4-4. Astrionic Module Reliability

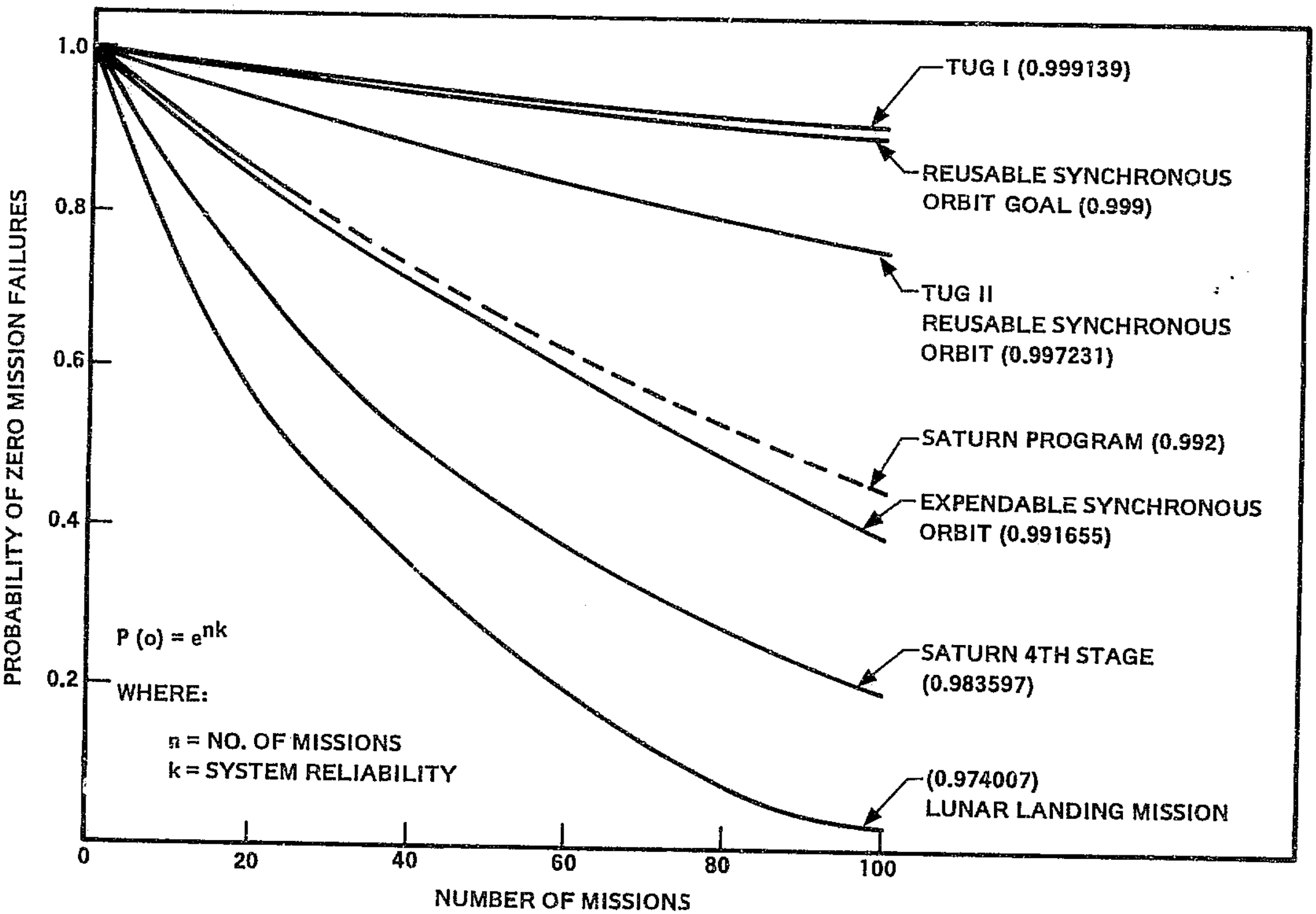

Figure 4-5. Space Tug Mission Reliability

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M-2. "Design Techniques for Modular Architecture for Reliable Computer Systems," W. C. Carter, et al, Report Number 7-0208-0002, IBM SSC, Huntsville, Alabama, March 26, 1970.

M-3. "A Non-Orthogonal Gyro Configuration," M. S. Thesis, MIT Department of Aeronautics and Astrionics, Cambridge, Mass. January 1967.

APPENDIX N

SPACE TUG SAFETY ASSURANCE ANALYSIS

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1.0 INTRODUCTION

This appendix presents the results of the space tug safety assurance analysis. Both unmanned and manned space tug missions are addressed. The safety assurance analysis is concerned with identifying those parameters of the propulsion, astrionic, crew and payload systems which should be monitored in order to identify off-nominal conditions which could cause loss of crew, vehicle, payload or mission. Further, the analysis is concerned with recommending emergency action, depending on mission phase, which should be taken if an off-nominal event occurs.

2.0 GUIDELINES AND GROUNDRULES

- It is assumed that the propulsion module will have one main engine.
- Crew, in the context that it is used in this appendix, includes depot, space station, RNS, and space shuttle as well as tug crews when these crews are in proximity to the tug.
- In the event of a catastrophic failure, a rescue mission can be performed by another tug.
- The crew module will be separable from the astrionic module/propulsion module.
- The safety assurance function will provide the following, as required:
  - Automatic, manual and remote abort modes.
  - Phase independent/dependent procedures for responding to catastrophic and critical malfunctions/conditions to assure crew and vehicle safety or completion of primary mission objective.
  - Caution and warning displays located in the crew module to inform crew of critical off-nominal conditions.
- The astrionic components will be rated as to their criticality as follows:
  - Criticality I failure is one that jeopardizes the safety of the crew.
  - Criticality II failure is one that causes a primary mission abort.
  - Criticality III failure is one that neither jeopardizes crew safety nor results in primary mission abort but causes other mission impact. (For example, a failure which causes loss of vehicle autonomy.)

3.0 SUMMARY OF RESULTS

3.1 CATASTROPHIC AND CRITICAL MALFUNCTIONS/CONDITIONS

A catastrophic malfunction or condition is one that could cause an immediate injury or damage hazard to the crew, vehicle or payload. A critical malfunction or condition, while not causing an immediate hazard, may prevent successful completion of the mission. The following malfunctions and conditions have been identified as catastrophic for the crew, vehicle or payload:

- Loss of Attitude Control
- Guidance Reference Fail
- Loss of Electrical Power
- Reactant Tanks Explosion Hazard
- Battery Explosion Hazard
- TVC Actuator Hardover or Stationary in One Position
- Loss of Crew Module Life Support System
- Loss of Crew Module Electrical Power

The following malfunctions and conditions have been identified as critical for the mission:

- Loss of Rendezvous and Docking Radar
- Loss of Temperature Control
- Loss of Landing Radar

The parameters necessary to detect the above malfunctions and conditions and the recommended emergency actions are discussed later in this appendix.

3.2 AUTOMATIC, MANUAL AND REMOTE MISSION ABORTS

A requirement to automatically separate the crew or cargo module from the vehicle in order to preserve their safety has not been identified except for the four stage Saturn V mission when it is manned. The emergency malfunction detection and response portion of the safety system will be an open loop system. The decision to separate could be made by the crew or at a remote location. The on-board computer could be programmed to make separation decisions for unmanned tug missions. An automatic sequence to shutoff the main engine will be required in the event of a hardover TVC actuator.

3.3 LOSS OF VEHICLE AUTONOMY

Loss of vehicle autonomy can occur with failures in the navigation function. This situation will require that the tug receive command and data assistance from the ground, space station, or nearby vehicles in order to perform the mission.

3.4 SAFETY ASSURANCE IMPACT ON PRIME SYSTEM DESIGN

- Capability should exist to separate the crew module or cargo module from the vehicle.
- Capability should exist to separate the crew module/astrionic module from the propulsion module (manned missions).

- Pyrotechnic devices are recommended as backup capability for a module interface locking mechanism failure to unlock.

4.0 DETAILED ANALYSIS

4.1 UNMANNED MISSIONS (REUSABLE)

The navigation, power generation, guidance and control, thermal control and structures have been identified as critical functions. Mission requirements, component criticality and critical parameters have been defined for these subsystems. Mission rules for responding to failures in these subsystems have been defined for the synchronous orbit missions, i.e., delivery and retrieval of satellites. No automatic abort requirement has been identified for unmanned missions.

4.1.1 Navigation Function

4.1.1.1 Mission Requirements

- Provide inertial reference, attitude, and velocity information for guidance for all flight phases.
- Provide for autonomous IMU alignment.
- Provide for autonomous navigation update (vehicle state update).
- Provide range, range rate, angular position, and angular rate for rendezvous and docking operations.

4.1.1.2 Component Criticality

The following paragraphs present the rationale for selecting navigation function component criticalities. These criticalities are based upon the impact of losing the component output without regard to redundancy. For example, when analyzing the IMU to determine its criticality, it was assumed that its output failed regardless of the number of individual failures required to accomplish this. Component criticality is summarized in Table 4-1.

Table 4-1. Navigation Function Component Criticality

Component	Criticality
IMU	II
Horizon Sensor or Landmark Sensor	III
Star Trackers	III
Rendezvous and Docking Radar	II

4.1.1.2.1 IMU Alignment and Navigation Update Sensors

An operational IMU is mandatory for all mission phases. Loss of either attitude or velocity information during any phase will result in loss of mission or vehicle. Periodic IMU alignment and vehicle navigation update may be required. Alignment is accomplished automatically by two star trackers which fix the position of the vehicle with respect to the inertial reference. If one star tracker fails, the other star tracker can satisfactorily align the IMU by taking one star fix at a time. Navigation update is accomplished automatically by the two star trackers and a horizon or landmark sensor. Again, a failed star tracker is not critical. One star tracker can be used. If the IMU has been aligned, only the horizon or landmark sensor is needed. Loss of horizon or landmark sensor will result in loss of vehicle autonomy. Ground support or support from another space element (e.g., space station) would be needed to perform the navigation update.

4.1.1.2.2 Rendezvous and Docking Radar

The rendezvous and docking radar provides for autonomous rendezvous and docking operations of the space tug. A failed radar would not necessarily preclude a rendezvous operation if navigation assistance is provided by ground support or another space element. Automatic docking cannot be accomplished with a failed radar. Remotely operated docking with a target that does not have a docking radar cannot be accomplished with a failed tug radar. Thus, loss of the radar will result in loss of those missions where it is essential that an automatic docking occur. Further study will be required to determine if remotely operated docking with a target that has a docking radar can be safely accomplished.

4.1.1.3 Critical Parameters

The navigation sensor outputs will normally be processed by the on-board computer. The computer can perform reasonableness checks on the data to identify a failed component and issue appropriate alert signals such as the following:

- Guidance Reference Fail
- Horizon or Landmark Sensor Fail (not critical, but requires navigation update via command link)
- Rendezvous and Docking Radar Fail
- Star Tracker Fail (not critical, but requires computer commands to operational Star Tracker to accomplish both star sightings)

These sensors may also have self-test capability to detect internal malfunctions. Failure signals appropriate to the internal malfunction will be made available to the computer which could issue the aforementioned alert signals.

4.1.2 Power Generation Function

4.1.2.1 Mission Requirement

Provide sufficient electrical power to operate all astrionic subsystems.

4.1.2.2 Component Criticality

The power generation system consists of two fuel cells which are normally sharing the total power load. If either fuel cell fails the other cell is capable of supporting the total power load. However, this situation will require power distribution reconfiguration to open circuit the bad cell and put the operational cell on both main buses. Further, it is assumed that an explosion hazard could exist due to the cryogenically stored consumables (oxygen and hydrogen). Thus, the fuel cells and their reactant supply systems are rated as Criticality II components.

4.1.2.3 Critical Parameters

The following parameters have been identified as critical and are the recommended minimum measurements to be taken by the onboard checkout function for malfunction detection:

- Fuel Cell Output Voltage
- Fuel Cell Output Current
- H_2 Tank Pressure
- O_2 Tank Pressure

The output voltage and current measurements will provide an indication of degraded fuel cell operation. The pressure measurements will provide an indication of reactant leakage or an impending explosion hazard.

4.1.3 Guidance and Control Function

4.1.3.1 Mission Requirements

- Issue engine start, stop, throttle and attitude commands to place the vehicle in a desired position and orientation in space.
- Maintain vehicle attitude control.

4.1.3.2 Component Criticality

With the possible exception of the four stage Saturn V mission, there are no unique guidance and control function components. These functions are performed by the on-board digital computer using data supplied by the navigation sensors and resulting in commands to the propulsion module engine and RCS thrusters. Control accelerometers and a rate gyro package may be required for the four stage Saturn V mission for load relief and vehicle stability during S-IC burn. The on-board digital computer is categorized as Criticality II.

4.1.3.3 Critical Parameters

4.1.3.3.1 Loss of Attitude Control

Loss of attitude control is a catastrophic failure which could be detected by the on-board computer by checking for the following:

- Excessive attitude errors or rate
- Sequencing malfunctions
- Loss of computational capability (computer self-check)

4.1.3.3.2 Loss of Engine Thrust

The parameter to monitor to detect this situation is the acceleration along the vehicle longitudinal axis. This information can be derived by the computer from IMU data and compared to a stored acceleration value as a function of mission time and thrust command.

4.1.3.3.3 TVC Actuator Failures

In order to detect actuator failures (hardover or stationary in one position) the engine position and the engine position command in both pitch and yaw should be compared for agreement within limits.

4.1.4 Thermal Control Function

4.1.4.1 Mission Requirement

Maintain an acceptable temperature environment for the astrionic equipment during both active and passive tug modes.

4.1.4.2 Component Criticality

The critical failure mode is the loss of temperature control which will result in overheating or overcooling of the astrionic equipment. The active system includes a coolant pump, accumulator and radiators, all of which, if failed, could cause loss of coolant temperature control. This would result in loss of astrionic equipment temperature control and jeopardize successful mission completion. Thus, the thermal control components are rated as Criticality II.

4.1.4.3 Critical Parameters

- Coolant temperature
- Coolant pump inlet and outlet pressures

The coolant temperature measurement will provide an early indication of loss of satisfactory control of coolant temperature. Temperature sensors on electrical components may also be monitored. The pump pressures will provide an indication of pump and accumulator operation.

4.1.5 Structures

The analysis of the structural system is limited to the module interface lock.

4.1.5.1 Mission Requirement

Physically lock or unlock adjacent modules after docking or prior to separation.

4.1.5.2 Component Criticality

Failure of the module interface locking hardware to unlock would prevent separation of modules. As will be seen in the mission rules, there are failures for which it is desirable to separate the payload from the vehicle in order to prevent possible damage to the payload. Further, a failure to separate may result in loss of mission. Thus the locking hardware is rated as a Criticality II device.

Since the locking hardware is a mission critical device, pyrotechnics are recommended as backup to separate the payload module from the vehicle.

4.1.5.3 Critical Parameters

The critical parameters for the locking hardware would be discrete signals (from limit switches) which would indicate when locking and unlocking is complete.

4.1.6 Mission Rules for Synchronous Orbit Missions (Reusable)

Table 4-2 outlines mission rules for response to catastrophic and critical malfunctions/conditions for the reusable synchronous orbit missions.

4.2 MANNED MISSIONS

The critical functions previously discussed in the unmanned missions section will also be critical for manned missions. The mission requirements and critical parameters which were defined for these functions will be the same as for unmanned missions. Component criticality for the navigation, guidance and control, and power functions will change due to the presence of the crew. Additional components are considered for manned missions. These are a landing radar and an emergency supply battery. These components and their respective component criticalities are discussed in the succeeding paragraphs. Further, the caution and warning displays, life support and CM electrical power systems are discussed. Finally, mission rules for responding to catastrophic and critical malfunctions/conditions have been defined for the lunar landing mission. Automatic abort for manned four stage Saturn V missions may be required for crew safety. No other automatic abort requirement has been identified for manned missions.

4.2.1 Navigation Function

In addition to the sensors previously discussed, a landing radar is included to provide for automatic lunar landing capability. The landing radar will provide altitude, descent rate and horizontal velocity for guidance and control.

Table 4-2. Synchronous Orbit Mission (Unmanned Reusable) Mission Rules

Malfunction/Condition	Phase	Ruling–Comments
Guidance Reference Fail	All	Abort Mission–Deactivate Vehicle
Loss of Attitude Control	All	Abort Mission–Deactivate Vehicle
Horizon or Landmark Sensor Fail	All	Continue Mission–Autonomy is lost. Achieve nav. updates via command link.
Loss of Star Tracker	All	Continue Mission–Computer commands other star tracker to make two star sightings
One Fuel Cell Failed	All	Continue Mission–Configure operational cell to both buses. Open circuit failed cell.
Both Fuel Cells Failed	All	Abort Mission–Deactivate Vehicle
Reactant Tank Explosion Hazard	All	Separate payload to safe distance. Continue to monitor tank pressure. If pressure drops to safe levels dock with payload and continue mission. (Decision could be made to continue mission with risk.)
Loss of Engine Thrust, Actuator Hardover or Stationary in One Position	Main Engine Burn	Automatic Engine Shutoff–If RCS can provide required velocity change to achieve sync orbit, complete mission and deactivate vehicle. If RCS can provide required velocity change to return to LEO, rendezvous and dock with shuttle.
	LEO	Switch to RCS control–rendezvous and dock with shuttle.
	Sync Orbit	Switch to RCS control–place payload and deactivate vehicle.
Loss of Coolant Temp. Control	LEO	Abort mission–rendezvous and dock with shuttle.
	Orbital Transfer and Sync Orbit	Complete mission–deactivate vehicle. (Equipment temperature tolerances are not immediately exceeded with loss of coolant temperature control. However, if time to go to achieve sync orbit is greater than predicted time for unacceptable temperature levels to occur, it may be desirable to abort the mission and rendezvous and dock with shuttle.)
Rendezvous and Docking Radar Fail	Rendezvous	Continue Mission–receive navigation data via command link.
	Docking	No go for docking.
Locking Hardware Unlock Fail	Separation	Fire Pyro Device
Locking Hardware Lock Fail	Docking	Abort Mission–command locking hardware to unlock position and separate.

4.2.1.1 Landing Radar Critical Parameters

The data provided by the landing radar will normally be processed by the on-board digital computer. As with the other navigation sensor outputs, the computer can perform reasonableness checks on the landing radar data to detect failures and then issue the alert signal-Landing Radar Fail.

4.2.1.2 Component Criticality

An IMU failure (Guidance Reference Fail) is a catastrophic failure. The safety of the crew is immediately jeopardized and may require that a rescue mission be performed. Loss of a star tracker or horizon or landmark tracker will prevent autonomous vehicle operation, but the mission can still be performed and crew safety is not jeopardized. The capability to automatically rendezvous and dock is not considered critical for mission performance. Rendezvous could be accomplished with navigation data supplied by the ground or another space element. Docking could be performed in a manual guidance mode. For lunar landing missions, an operational landing radar is required until an adequate navigation update is accomplished after which the crew could manually guide to the landing site. A failed radar prior to the update would require a mission abort. The navigation function component criticality is summarized in Table 4-3.

4.2.2 Guidance and Control

The onboard digital computer is a Criticality I component for manned missions.

4.2.3 Power Function (Astrionic Module)

The primary power for manned missions is similar to that for unmanned missions. The manned mission configuration will include an emergency supply battery which will be used only if the primary power system fails, and then it will provide loads for only the critical components.

4.2.3.1 Emergency Supply Battery Critical Parameters

It is possible for an explosion hazard to develop with batteries. This can occur if the battery becomes overcharged or has a reverse charging cell. Both of these conditions cause the release of gases in the sealed cells and subsequent increase in pressure. A reverse charging cell can be detected by a battery voltage measurement. An overcharged cell can be detected by a battery temperature measurement.

Table 4-3. Navigation Subsystem Component Criticality

Component	Criticality
IMU	I
Horizon or Landmark Sensor	III
Star Trackers	III
Rendezvous and Docking Radar	III
Landing Radar	II

Table 4-4. Manned Lunar Landing Mission Rules

Malfunction/Condition	Phase	Ruling–Comments
Loss of CM Life Support System	All	Abort Mission–Switch to emergency system or supply and dock with LOSS ASAP. (On lunar surface, it may be desirable to place as low a demand on the backup life support system as possible and await rescue.)
Loss of CM Electrical Power	All	Abort Mission–Switch to emergency supply and dock with LOSS ASAP.
Guidance Reference Fail or Loss of Attitude Control	Lunar Orbit	Abort Mission–Separate CM/AM from PM. Place CM in manual attitude control mode and await rescue.
	Powered Descent and Ascent	Abort Mission–Attempt to achieve orbit. Separate CM/AM from PM. Place CM in manual attitude control mode and await rescue. At some time during powered descent, e.g., landing site visible, it may be desirable to land and await rescue rather than attempt to abort to orbit. Both abort modes will require manual guidance takeover.
Horizon or Landmark Sensor Fail	All	Continue Mission–Autonomy is lost. Achieve nav. update via command link or manually via optical alignment.
Loss of Star Tracker	All	Continue Mission–Computer commands other star tracker to make two star sightings or manually via optical alignment.
One Fuel Cell Failed	All	Continue Mission–Configure operational cell to both buses. Open circuit failed cell.
Both Fuel Cells Failed	All	Abort Mission–Dock with LOSS ASAP. The emergency supply battery should be configured to the critical load bus and both fuel cells open circuited.
Reactant Tank or Battery Explosion Hazard (AM)	All	Abort Mission–Separate CM from AM/PM as soon as practical. Place CM in manual attitude control mode and await rescue. If on lunar surface, AM shall be powered down and crew will take appropriate safety precautions.
Loss of Engine Thrust	Lunar Orbit	Abort Mission–Switch to RCS control, rendezvous and dock with LOSS.
	Powered Descent and Ascent	Abort Mission–Using RCS attempt to remain in or achieve some nominal orbit and await rescue.
Actuator Hardover or Stationary in One Position	Lunar Orbit	Abort Mission–Switch to RCS control, rendezvous and dock with LOSS.
	Powered Descent and Ascent	Automatic Engine Shutoff–Attempt to remain in or achieve some nominal orbit using RCS and await rescue.
Loss of Coolant Temp. Control	Lunar Orbit	Abort Mission–Rendezvous and dock with LOSS.
	Powered Descent and Ascent	Continue Mission
Rendezvous and Docking Radar Fail	All	Continue Mission–Navigation assistance required from LOSS for rendezvous. Manual control takeover required for docking operations.
Landing Radar Fail	Lunar Orbit	Abort Mission–Rendezvous and dock with LOSS.
	Powered Descent	Continue Mission–If adequate update of vehicle state vectors has been accomplished. Otherwise abort mission, rendezvous and dock with LOSS.

4.2.3.2 Component Criticality

A catastrophic failure mode exists with both the battery and the fuel cell. A battery or reactant tank explosion would jeopardize crew safety. Thus, the power subsystem components are rated as Criticality I.

4.2.4 Caution and Warning Displays

The caution and warning capability will be provided by the crew display function. A critical off-nominal condition will be displayed, and an audible tone will be generated in the crew module. (See Crew Display, Appendix O.)

4.2.5 Crew Module Life Support and Electrical Power Systems

Failures in the crew module life support and electrical power systems may immediately jeopardize crew safety. The critical parameters necessary to detect catastrophic and critical malfunctions/conditions would be displayed to the crew. The crew, aided by mission rules dedicated to these two systems, would decide what response is required for a catastrophic or critical malfunction/condition.

It is possible that the astrionics will provide primary electrical power to operate the crew module life support system. If so, then it would be desirable to have the crew and astrionic modules separable from the propulsion module. The crew could then separate from the vehicle, whenever required for their safety, and retain the primary power source for their life support system. However, the occurrence of an explosion hazard in the AM primary power system will require that the crew module separate from the vehicle and operate the CM life support system with the CM emergency power supply.

4.2.6 Mission Rules for Lunar Landing Missions

Table 4-4 outlines mission rules for response to catastrophic and critical malfunctions/conditions for lunar landing missions. It is assumed that a Lunar Orbiting Space Station (LOSS) is present.

REFERENCES

N-1 Final Flight Mission Rules Apollo 13 (AS-508/109/LM-7) MSC-01807, February 12, 1970

APPENDIX O

SPACE TUG DISPLAY CONSIDERATIONS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

The purpose of this study is to address displays and controls required by the space tug crew to perform assigned missions. The two types of displays and controls that will be presented are:

- Switches, dials, lights, etc.
- Multipurpose Electronic Displays (MED)

Major emphasis will be placed on the MED.

The philosophy for manned missions is to have a crew module as part of the space tug. The three man tug crew is composed of:

- Commander
- Pilot
- Experimenter

The commander and pilot will each have a flight station and the experimenter a "housekeeping" station. The commander and pilot stations will each have a multipurpose electronic display (MED) that will present command and status information. The third station will have a status MED. The space tug will have the capability of being operated by either the commander or pilot. The flight crew will have the capability to select what MED information is to be displayed by keying the select button on the keyboard.

The information will be presented as command or status. It will be presented:

- On call by the operator
- When an out-of-tolerance condition exists.

In addition to the MED and keyboard the two pilots will have a manual control stick. The manual control stick will have manual override capability; thereby, the flight crew will have the option of switching from an automatic sequence to manual by keying in the code.

2.0 STUDY GUIDELINES AND GROUNDRULES

The following guidelines and groundrules form the basis for the displays and controls:

- Docking and lunar landing may be completely automatic
- Be capable of being flown by one crewman
- Be operated automatically
- Various types of displays and controls will be considered
- Manual control override possible

3.0 SUMMARY OF RESULTS

Displays and controls as used in the Apollo command module and lunar module such as meters, switches, dials, etc. were considered. This method for display and control for the space tug was rejected for the following reasons:

- Requires complex electrical design to provide automatic checkout, operations and onboard autonomy
- Requires constant monitoring by the pilots
- Requires more cabin space than centralized MED Display concept.

With the current emphases placed on autonomy in space vehicles, use of multipurpose electronic displays and keyboards is favored.

The Cathode Ray Tube (CRT) was chosen as the MED for the space tug in this study for the following reasons:

- State-of-the-art
- Currently planned for other space transportation systems.

A follow-on trade study of the following advanced displays is recommended:

- Deformagraphic Storage Display Tube (DSDT)
- Light Emitting Diode (LED)
- Fiber Optics
- Others

A circuit breaker panel has been allocated for instantaneous access for power on/off.

3.1 DISPLAY AND CONTROL DEFINITION

The display and control system is the method the space tug crew utilizes to monitor, maneuver, and perform other mission operations while in a space environment.

3.2 DISPLAY AND CONTROL CHARACTERISTICS

The physical characteristics of the displays and controls considered are:

- Weight
- Power
- Heat dissipated

The physical characteristics of the cathode ray tube, keyboard and manual control signal converter are given in Table 3-1.

Table 3-1. Display and Control Characteristics

Unit (Total)		Wt (Lbs) (Ea.)	Pwr (W) (Ea.)	Heat Dissipation (BTU/Hr) (Ea.)
CRT (7 x 9)	(3)	25	180	380
Keyboard	(3)	13	20	TBD
Manual Control Signal Converter	(2)	40	120	TBD

4.0 DETAILED ANALYSIS

4.1 DISPLAY REQUIREMENTS

4.1.1 Display and Control System

The display and control system will be comprised of the following elements:

- Commander's display
- Commander's keyboard
- Commander's manual control stick
- Pilot's display
- Pilot's keyboard
- Pilot's manual control stick
- Experimenter's display
- Experimenter's keyboard
- Communication panel

4.1.2 Display and Control Display Format

Flight commands and status information will be presented to the crew as formatted information. The information to be displayed will be presented on the MED that has been preselected by keying the information through the keyboard. The information will be presented in the manner described below. The formats shown are only selective examples of a few parameters to be monitored. In a flight presentation, several parameters would appear on the MED simultaneously.

4.1.2.1 Altitude, Reference Figure 4-1

Altitude is indicated on a vertical fixed scale. The thousands of feet are indicated by numerals below the vertical scale. The hundreds are indicated on a scale with major increments every 10 feet and minor increments every 2 feet. The altitude value will be indicated by a moving symbol (<). The command altitude will be displayed by the symbol (=).

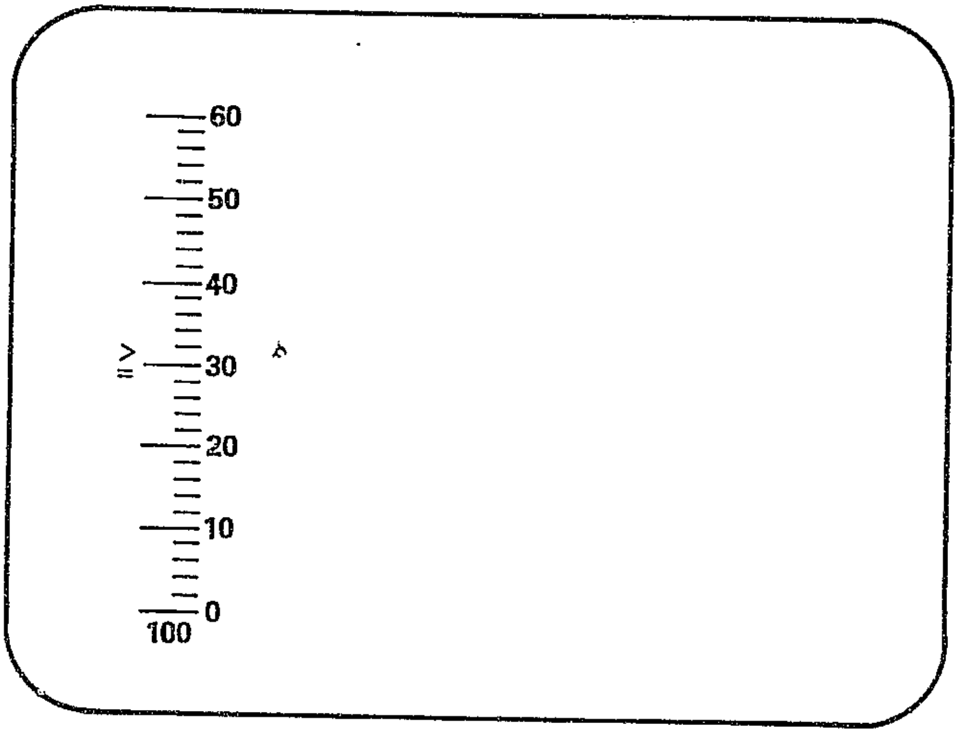


Figure 4-1. Altitude Display Format

When the commanded and actual altitude are the same, the symbol will be superimposed.

4.1.2.2 Angle of Attack, Reference Figure 4-2

The angle of attack is indicated on a vertical fixed scale with 5° major increments and 1° minor increment. The scale limits will be from -10° to $+20^{\circ}$. The angle of attack index symbol (<) moves vertically with reference to the center line of the vehicle.

4.1.2.3 Apogee/Perigee, Reference Figure 4-3

This flight parameter will be presented to the pilot as a decimal readout. The display will be indicated to the nearest 10th of a nautical mile.

4.1.2.4 Check List Data (Data Bus), Reference Figure 4-4

The unit with the specific measurements will be displayed. If an out-of-tolerance condition exists, the symbol (*) will appear to the right of the unit name.

4.1.2.5 ΔV Remaining (fps), Reference Figure 4-5

The digital readout of ΔV presents data throughout the maneuver. The vertical scale with indicator (>) provides data for the last 50 fps.

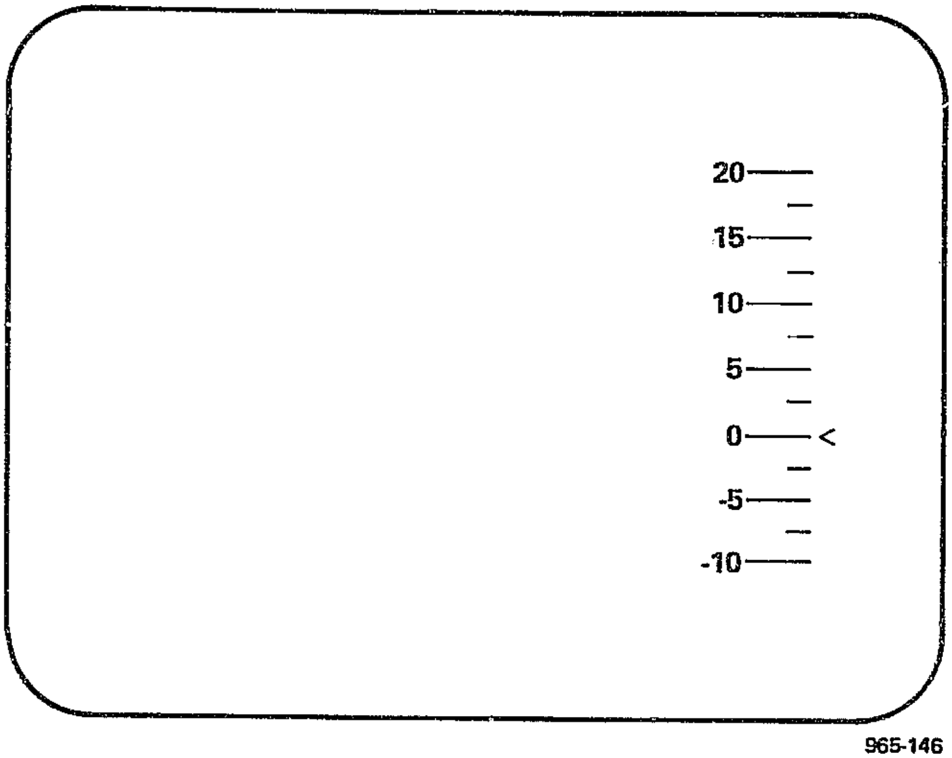


Figure 4-2. Angle of Attack Display Format

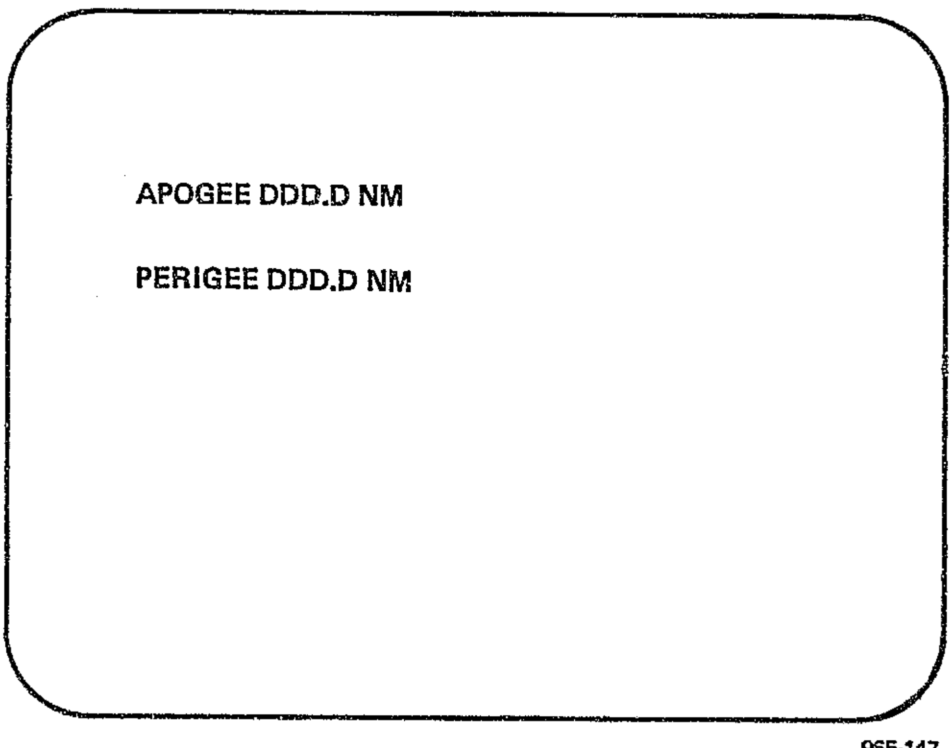
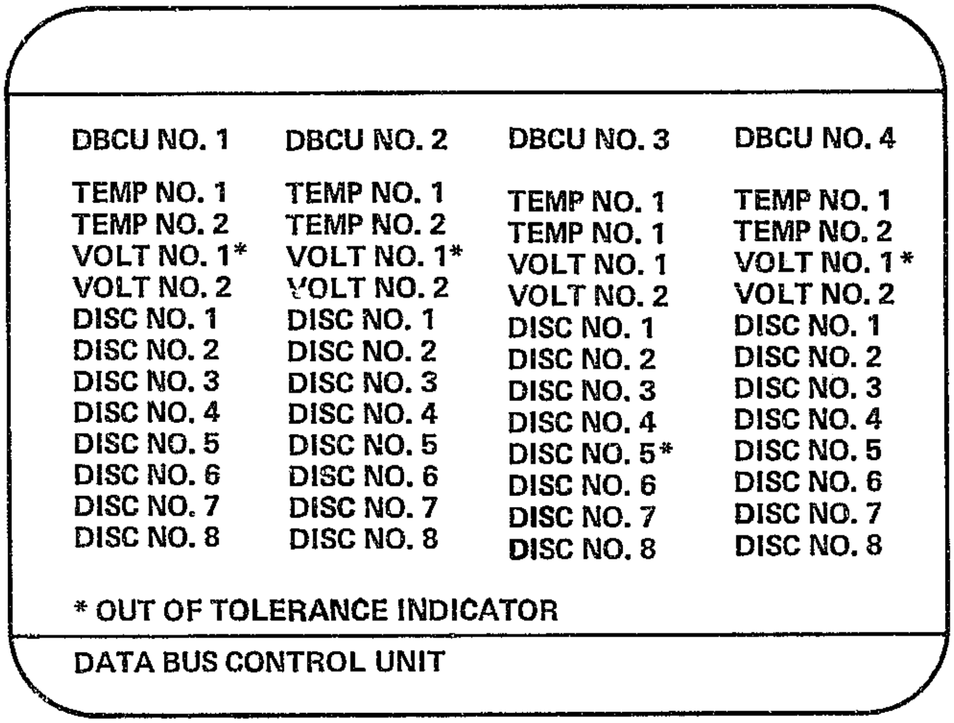
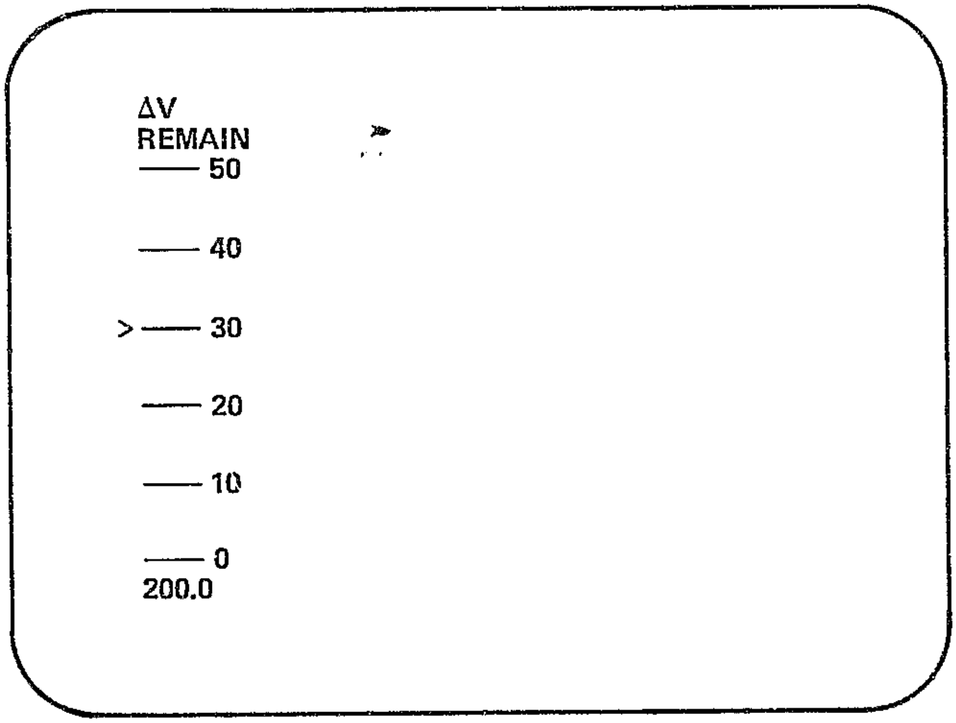


Figure 4-3. Apogee/Perigee Display Format



965-148

Figure 4-4. Check List (Data Bus Control Unit) Display Format



965-149

Figure 4-5. ΔV Remaining (FPS) Display Format

4.1.2.6 Docking Grid, Reference Figure 4-6

A fixed grid with lines representing vehicle XY and XZ planes is presented. The scaling will be in feet with major increments of 10 feet and minor increments of 2.5 feet. The maximum distance displayed will be 50 feet. Other display formats will be used prior to this distance. The target will be depicted as a moving symbol (-□-). When the target symbol is positioned in the grid, the space tug will be in the correct attitude for docking.

Also included on the docking grid format is the range and range rate relative to docking. The range scale will be a decimal readout, reading from 0.1 feet to 100 NM. The rate of approach will also be a decimal readout, reading from 0.1 fps to 100 fps.

4.1.3 Display Keyboard, Reference Figure 4-7

To send commands, verify commands, and monitor the status of the various parameters, a 30-key keyboard is provided.

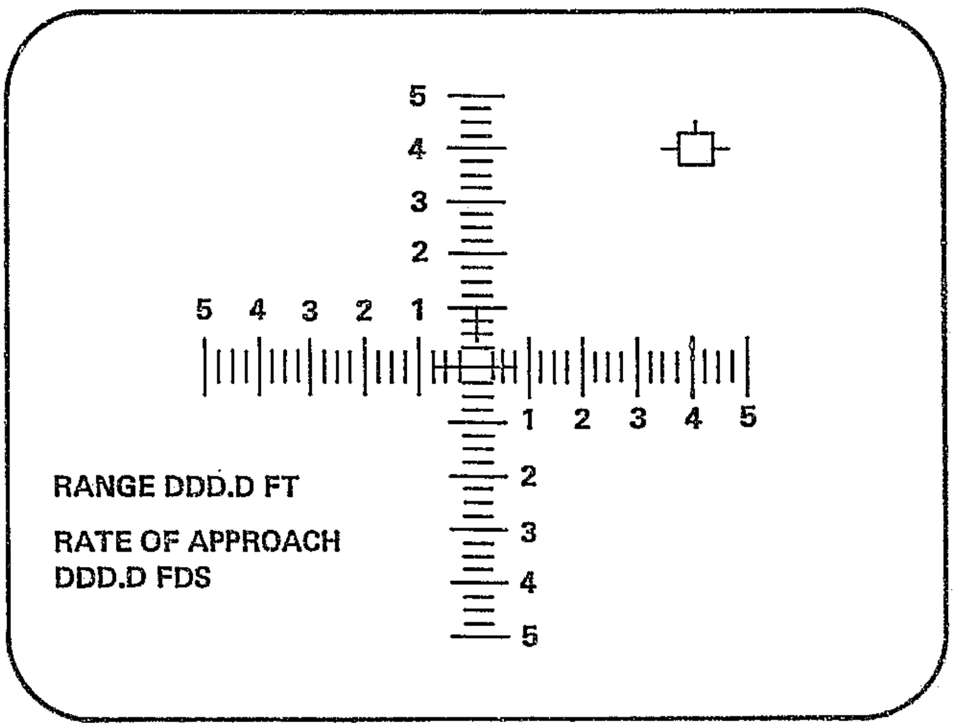


Figure 4-6. Docking Grid Display Format

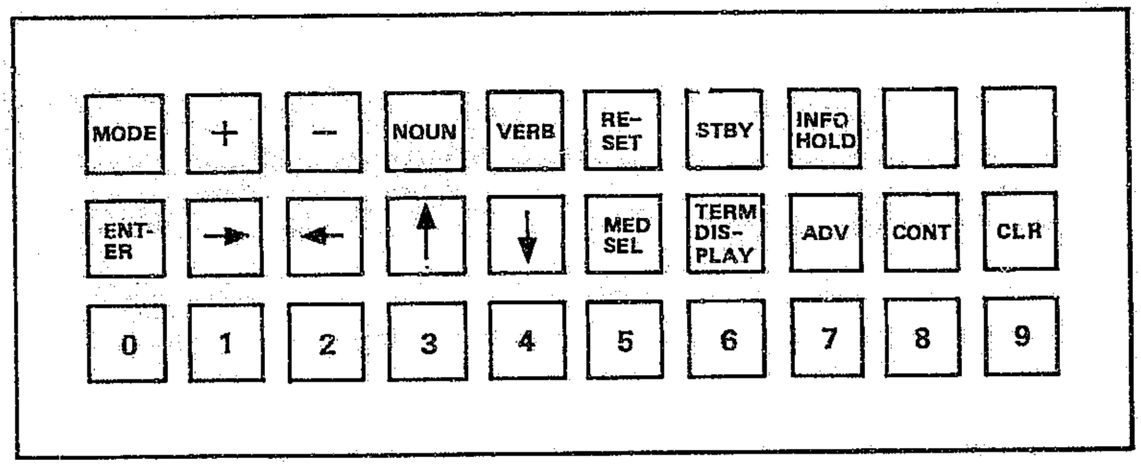


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Figure 4-7. Space Tug Keyboard

An explanation of the keyboard follows:

MODE – Used to select the flight mode vehicle is in and thereby select display parameter set.

+AND – Are sign keys used to identify the decimal data.

NOUN – Pushing this button indicates that the next two numerical characters keyed are to be interpreted as the noun code.

VERB – Pushing this button indicates that the next two numerical characters keyed are to be interpreted as the verb code.

STANDBY – Is keyed to stop computer to hold for update

INFORMATION HOLD – To hold information on CRT longer than normal time.

RESET – Turn off alarm conditions on MED providing the alarm condition has been corrected.

CLEAR – Used during a data loading sequence to clear or blank the data display register being used. It allows the operator to reload the data word.

ENTER – Is used as follows:

- To direct or execute the verb/noun code
- To accept a data word just loaded
- In response to a "please perform" request

↑↓ ⇄ Used to locate cursor

MED SELECT – Used to select MED for displaying information

TERMINATE DISPLAY – Stop information being displayed to perform another operation.

ADVANCE – Used to advance the computer to another step.

CONTINUE – Used to pick up where the computer operation was stopped and proceed.

0-9 – Numeric numbers

Two keys are not used for this keyboard, providing growth capability. If more than two additional keys are required, another keyboard with more key positions would be used.

4.1.3 Manual Flight Control

The crew has the option of manual flight control. Manual control of the space tug will be provided by a side arm controller. Signal conversion will be through a flight control signal converter.

4.1.4 Display Console, Reference Figure 4-8

The display console configuration should be laid out for ease of operation and viewing for the crew. Crew viewing distance should not exceed 30 inches. However, it may be necessary to view 48 inches. Crew reach should not exceed 26 inches.

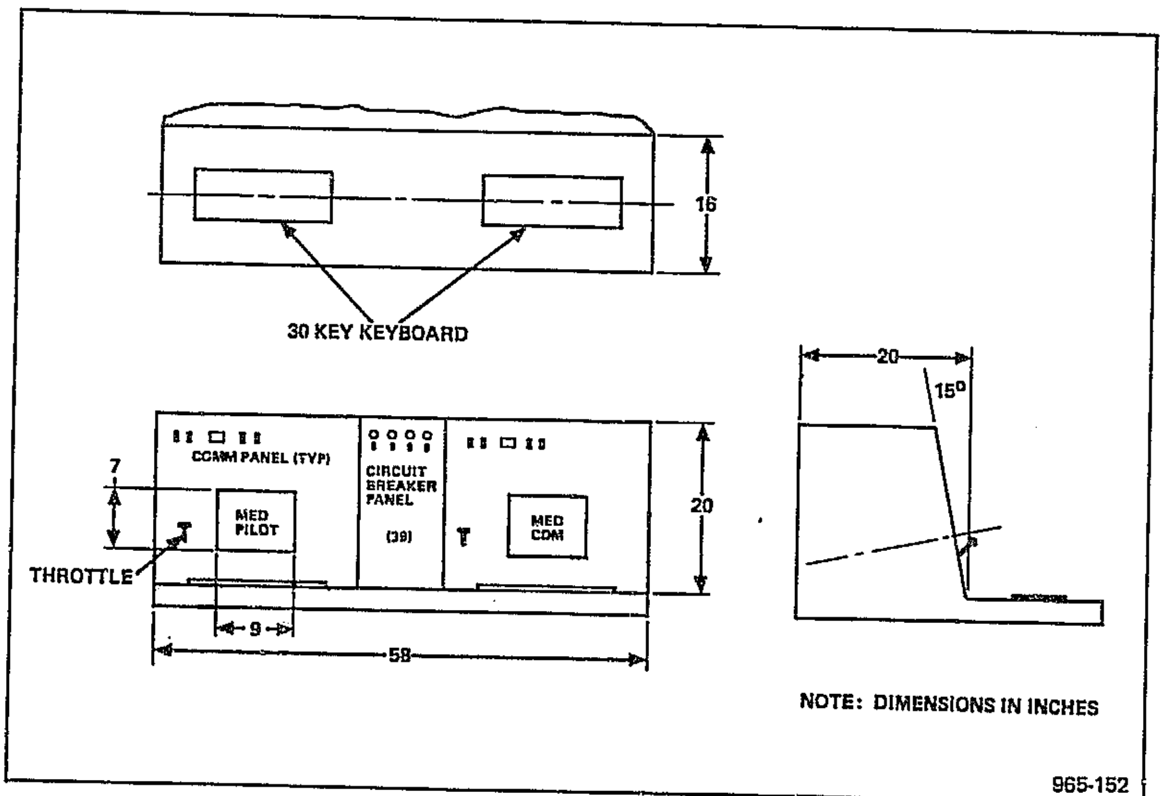

Figure 4-8. Space Tug Display Console

4.1.4.1 Display Characteristics

The characteristics of the display shall be as follows:

- The resolution of the display shall be a minimum of 512 line parts.
- The maximum effective spot size for stroke writing shall be 0.009 inches.
- The display reflectance shall not be greater than 10% of the ambient light.
- Contrast ratio shall be a minimum of 2.5 to 1 in a 1000 ft. lambert ambient with filters.
- The display refresh rate shall be 30 to 40 cps.
- Caution and warning shall be displayed at a rate of 1 to 4 times per second.
- Usable display area shall be a minimum of 7 x 9 inches.
- Character size - nominal 0.25 inch height, 0.18 inch width.
- Horizontal spacing between characters shall be a nominal 0.090.
- Vertical spacing between characters shall be a nominal 0.107.
- Display shall have 10-20 ft. lamberts of luminance after passing through any filters and with a minimum 2.5:1 contrast ratio.
- Filter design shall include dark (night) adaptation.
- The time required to display 16 lines, 32 characters per line shall not exceed 10 milliseconds.
- Control of display focus and intensity shall be provided.

4.1.4.2 Operational Environment

The display console shall have the following characteristics:

- Control panel illumination nominally 50-foot candles.
- Ambient illumination from a variable light source capable of 200 ft. candles.

4.1.4.3 Keyboard

The numeric keyboard shall have 30 keys and provide for a convenient crew interface.

- Keys shall be a minimum of 0.75 inches across, spaced on 1 inch centers.
- Key resistance shall be 5-10 ounces.
- Lettering shall be a minimum of 0.18 inches in height.
- Response to key signal is less than 50 milliseconds.

4.1.4.4 Circuit Breaker Panel

The circuit breaker panel shall be laid out with circuit breakers and a light. The light shall light up on an open in the circuit. A circuit breaker panel with 39 circuit breakers has been allocated for instantaneous access for power on/off.

4.2 CAUTION AND WARNING (C&W)

The caution and warning shall be displayed on the MED. The C&W will be superimposed on the MED and will be shown by cycling at a rate of 1 to 4 CPS. Audible signals will sound in the event that a warning signal is presented.

4.3 MISSION CONSTRAINTS

The crew display and control system allows one crew member to operate the space tug in automatic and/or manual modes during orbital coast/burn, rendezvous, docking, lunar landing, etc.

4.4 MISSION MODES

During the space tug stay in a space environment, it will operate in specified modes. The mission modes and their numeric designation listed are for lunar landing, but would be applicable to all manned missions by selecting the desired mission modes:

1. Reactivation
2. Undocking
3. Transfer Burn
4. Orbital Coast
5. Power Descent Landing
6. Deactivation
7. Storage
8. Reactivation
9. Powered Ascent
10. Orbital Coast
11. Rendezvous
12. Docking
13. Deactivation
14. Storage
15. Maintenance
16. Refueling

4.5 DISPLAY AND CONTROL INTERFACE

The displays and controls will interface with the astrionic system data bus as shown in Figure 4-9.

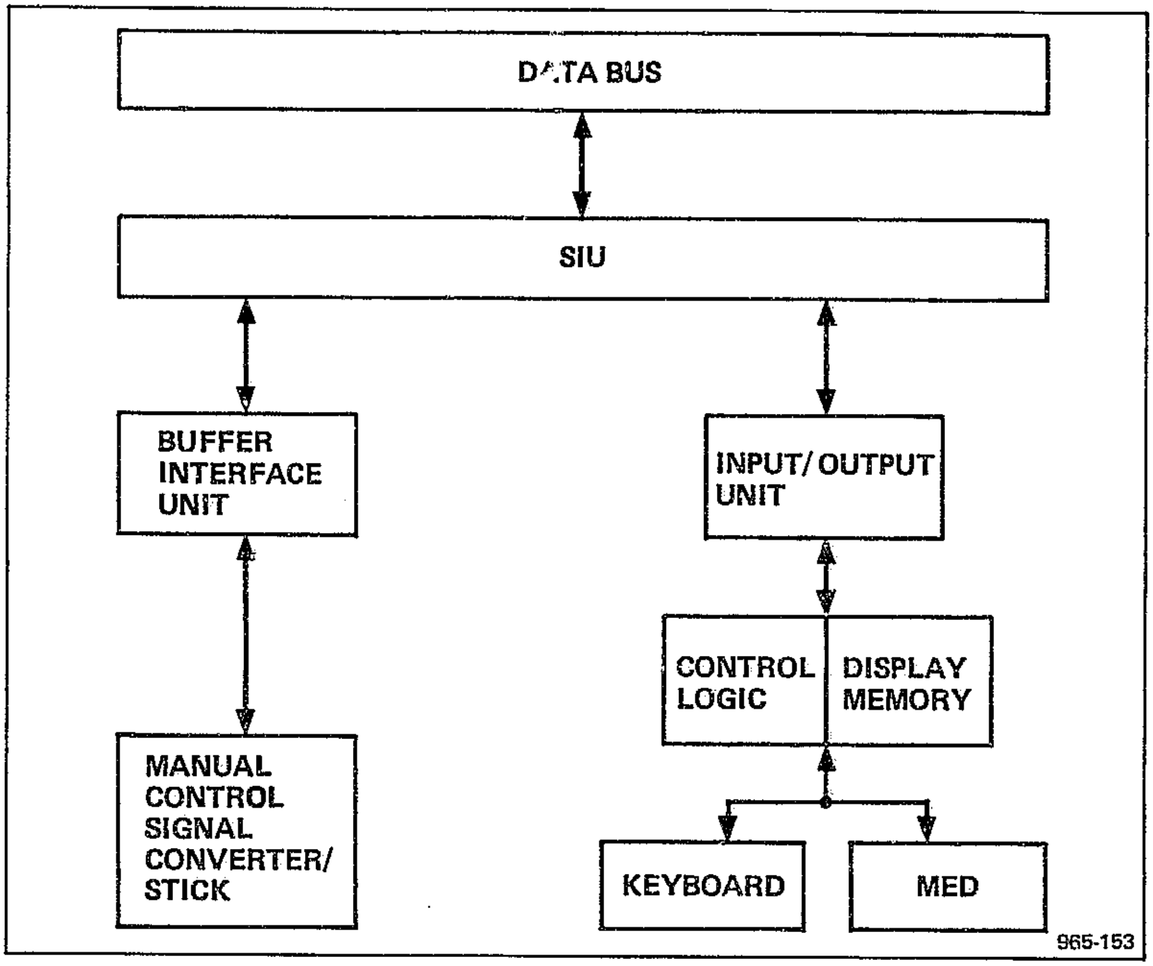

Figure 4-9. Display Interface

REFERENCES

O-1 "Preliminary Integrated Displays and Controls Subsystem Requirements Specifications," 1 June 1970, F. L. Holmes.

O-2 "Apollo Operational Handbook Block II Spacecraft," 15 April 1967, NASA.

O-3 "Apollo 12 AC Electronics," AC Div. of GM.

O-4 "Military Standard – Human Engineering Design Criteria for Military Systems, Equipment and Facilities," MIL-STD-1472, 9 February 1968.

O-5 "MSFC NASA Standard Human Design Criteria," MSFC-STD-267A, September 23, 1966.

APPENDIX P

RADIATION EFFECTS IMPACT ON SPACE TUG ASTRIONICS

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

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1.0 INTRODUCTION

The space tug will be exposed to both natural and induced radiation environments. This section discusses these environments; their effects on components, circuits, and systems; radiation hardening techniques; a system hardening plan; and the penalties associated with radiation hardening. Since this subject is presented in an unclassified document, many aspects are treated in general terms.

2.0 GROUNDRULES AND ASSUMPTIONS

No specific radiation levels, technology or system are discussed in this section since detailed specifications and definitions are not available, and an unclassified presentation is given. However, the data presented does include the radiation types and range of levels that should encompass any space tug radiation environment and radiation effects and hardening for components, circuit and system technologies representative of those that could be used to implement space tug astrionics.

3.0 SUMMARY OF RESULTS

The space tug may be exposed to both natural and induced radiation environments as follows:

Natural:

- Solar Cosmic Particles and Flares
- Natural Radiation Belt (Van Allen)
- Galactic or Extra-Galactic Cosmic Particles

Induced:

- Nuclear Reactor Radiation
  - NERVA
  - Space Base Power Reactors
- Pulsed Radiation (including artificial radiation belts)

For space tug missions now being considered, natural radiation environments have little impact on the design of space tug astrionics. However, in the case of energetic solar flare activity, space tug missions may have to be delayed or terminated early, thus imposing an operational constraint.

Interaction of the space tug with induced nuclear reactor radiation from NERVA and space base power reactors will impose minimum impact on tug astrionics design. Here again, however, tug operational safety constraints will have to be met in order to keep the tug in zones of low radiation environments.

By far, the most stringent environment to which the tug could be exposed is that from pulsed radiation. Techniques exist to harden electronics for this type environment but will impact cost, weight, power requirements, etc. These hardening techniques must be considered in the initial design of the space tug astrionic module. To harden a system once the system approach and technologies to implement that approach have been finalized becomes a formidable task.

4.0 DETAILED ANALYSIS

4.1 RADIATION ENVIRONMENT

In addition to the temperature, shock, and vibration environments associated with space systems, the space tug astrionic module must be designed to function properly in natural and induced radiation environments. The combined radiation environments present a multiplicity of types of radiation and effects on electronic subsystems. This section defines the radiation environments relevant to a space tug mission profile. The radiation environment presented by pulsed radiation and its interaction with the geomagnetosphere can be extremely hostile. The radiation levels to which the system is hardened depends upon whether the system is to be manned or unmanned at the time of and subsequent to the encounter. The electronic and mechanical components of the system can be designed to withstand much more severe doses than those lethal to a human biological system. There is a considerable difference in the cost of hardening a system from an equipment standpoint as against hardening to the lethal doses in tissue. Consequently, a distinction will be made between a manned (low level) or unmanned (high level) system radiation environment in the following sections.

4.1.1 Pulsed Radiation

The worst case, and most likely, pulsed radiation encounter is one occurring at high altitude (on the order of 50 to 300 nmi) near the end of the system lifetime. The radiative emanations of pulsed radiation at about 100 nmi have the general time history indicated in Figure 4-1.

The energy spectra and intensities of the various types of radiation are critically dependent on the radiation source, source altitude and range to the receiver. As a rule, the worst case is assumed for the hardening of a system. The range can be selected to be consistent with the maximum tolerable effect associated with one or more of the radiation types.

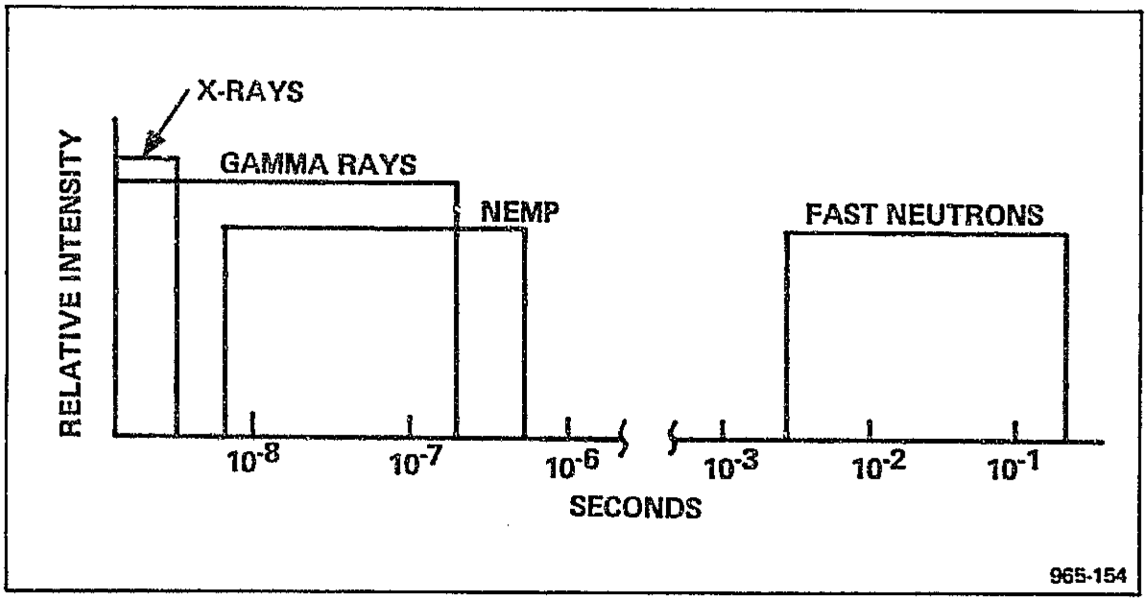

Figure 4-1. Idealized Pulsed Radiation Time History

4.1.1.1 Prompt Gamma Rays

Fission gamma rays have a spectrum which gives them the characteristic of being highly penetrating. It is futile to consider shielding on a system where weight reduction has a high priority.

The significant characteristics of prompt gamma rays on electronics are the ionization does rate [rads (Si)/sec] and the ionized dose [rad (Si)] discussed in Section 4.2.1.

4.1.1.2 X-Rays

The energy spectra of pulsed radiation X-rays permit the consideration of shielding as a hardening measure, even in a weight conscious system. The principal effects of X-rays are ionization, thermal, and mechanical as listed in Sections 4.2.1, 4.2.3, and 4.2.4, respectively.

4.1.1.3 Fast Neutrons

The neutron spectra associated with pulsed radiation are of sufficiently high energies that shielding is impractical for all but ground systems. The effects on electronics of low to moderate neutron fluences (10^{11} to 10^{13} n/cm^2) fall in the displacement damage category only (Section 4.2.2). The effects of high fluences (10^{13} to 5 x 10^{14} n/cm^2) include ionization as well as displacement damage. The passage of a high intensity pulse of fast neutrons through the materials of a system generates ionizing radiation which can contribute the major portion of the ionizing dose absorbed in components and materials. The nature of that dose is dependent on the choice of system materials. This effect is not significant for a system designed to tolerate doses only up to lethal doses in tissue. The X-ray, gamma ray, and neutron induced doses involving a high neutron fluence would be far in excess of the lethal dose range. A manned system is therefore immune by definition to one of the more serious problems confronting the design of a hardened unmanned system. If the space tug is designed to tolerate levels commensurate with an unmanned system, this effect will have a significant impact on total system design.

4.1.1.4 Nuclear Electromagnetic Pulse (NEMP)

Pulsed radiation is accompanied by two principal types of electromagnetic effects (References P-3, P-4, and P-5). (Electromagnetic radiation as used in this section applies to radiation of longer wavelengths than are characteristic of gamma and X-rays.) One involves alterations to the electrical properties of the atmosphere causing communication and radar blackout. This mechanism will not be considered in this study since only communications and radar in the vicinity of the pulsed source will be altered. The worst that could happen would be a local blackout in communication with the space tug from the ground which could last from minutes to hours. There would be no similar blackout associated with space-to-space communications (as opposed to ground-to-space or space-to-ground). The second electromagnetic effect will be of primary interest since it produces an electromagnetic pulse of short duration, possibly causing detrimental effects in the space tug astrionics.

There appears to be at least four different mechanisms whereby an electromagnetic pulse may produce problems in the astrionics from pulsed sources (References P-5 and P-9). The first two are associated with gamma and X-radiations producing asymmetric charge distributions in the region surrounding the source (Reference P-1). The third is the result of the rapid expansion of the essentially perfectly conducting plasma of residues in the earth's magnetic field (Reference P-2). Finally, gamma interactions in an asymmetric case will produce scattered electrons generating the electromagnetic pulse (Reference P-1). These mechanisms and the threats they produce will be discussed in the following sections. Since this paper is unclassified, the threat scenarios cannot be given in detail as to source yield, field strengths, time profiles, etc.

The first two mechanisms, often called the Compton-Electron (Reference P-5) and Photo-Electron (Reference P-14) Models are believed to be the principal means for generation of the NEMP from sources on or slightly above the earth's surface and near the top of the sensible atmosphere. The third mechanism called the Field-Displacement Model (Reference P-5) is responsible for the NEMP produced at greater altitudes where the only interacting medium is the geomagnetic field. Likewise, the fourth mechanism, the Gamma-Weapon Case Model (Reference P-14), is produced at great altitudes where the only media for interaction is either the pulsed source case or the geomagnetic field.

4.1.1.4.1 Compton-Electron Model. The initial gamma radiation leaves the source with very high energies colliding with electrons in the atoms and the molecules of the surrounding air, thus transferring energy (Reference P-5). These electrons generally move out rapidly from the center of the source. In order for these electrons to produce an electromagnetic pulse, some sort of asymmetrical medium must be available for interaction. This medium could be the earth, the atmospheric density gradient, or any structural material. These asymmetries produce an effective compton-electron pulse which is like a vertical radiating dipole. The current pulse in the air radiates electromagnetic energy just as it would if it were flowing in a wire transmitting-antenna, and this radiation constitutes the first part of the characteristic signal of the pulsed radiation. The compton-electrons, flowing away from the positive ions, produce a radial electric field. Under the influence of the electric field, there is a large number of electrons present which flow back toward the originating point. This initiates a second pulse of current which is rapidly terminated by recombination. This characteristic action occurs very rapidly, thus producing frequencies as high as one GHz. However, only a very small part of the total electromagnetic energy is radiated at such high frequencies. Most of the energy is centered around frequencies of tens to hundreds of kHz when detected at hundreds of miles. The possibility of a second NEMP produced by X-radiation will be discussed in the following section.

4.1.1.4.2 Photo-Electron Model. The mechanism is essentially the same as described above except the radiation producing the electrons is X-radiation instead of gamma radiation (Reference P-14). The interaction of gamma and X-radiation with matter is well documented and, thus, will not be discussed here. The photo-electron mechanism is most pronounced by pulsed radiation sources at or above an altitude of fifty miles. This mechanism produces the most intense electromagnetic signal immediately above the D-layer where the X-rays interact. The longitudinal electric field produced is of the order of a few thousand volts per meter. The transverse field is of the order of tens of volts per meter. Thus, the field strengths produced by X-radiation tend to be smaller than fields produced by gamma radiation. However, the photo-electron mechanism can be important where the electron yield is much greater than the compton-electron process described here.

4.1.1.4.3 Field-Displacement Model. This mechanism will be of significance when the space tug is in a synchronous orbit or in some extremely high altitude orbit (Reference P-5). Immediately after initiation of the pulsed radiation, the hot debris is essentially a highly ionized plasma which is expanding rapidly. A property possessed by all plasmas is a tendency to exclude a magnetic field such as that of the earth from its interior. The expanding plasma of residues thus causes a violent distortion of the earth's magnetic field. As a result of the interaction, an electromagnetic pulse is propagated over very long distances at high altitudes.

4.1.1.4.4 Gamma-Weapon Case Model. The gamma radiation from pulsed sources is never fully symmetric because of the presence of ancillary apparatus, external structures, or the carrying vehicle (Reference P-14). The polarization of radiation from this mechanism is random, whereas the natural asymmetries tend to be vertically polarized. There are at least two possible ways by which such an asymmetrical distribution of electrons might be generated: (Reference P-1) because of the design of the source, and (Reference P-2) because the shield material may be asymmetrically placed around the source. Such shields are to reduce the flux radiated in a given direction and thus make more difficult long-range detection of X-rays. The shield might be a flat plate or a hemisphere a few meters from the device so oriented as to shadow the source from possible X-ray detectors. This mechanism is normally of interest only above 10,000 miles.

4.1.1.5 Artificial Radiation Belts

A high altitude source of pulsed radiation can cause an artificial belt of electrons and protons to form, overlapping the Van Allen belt, within a few minutes to hours after initiation. The effect is to significantly enhance the radiation hazards in the region of the belts. The artificial proton flux is relatively inconsequential to the electron flux which has a fission beta decay spectrum. The latter is somewhat more energetic than the natural electron belt spectrum and has therefore greater penetrability and presents a more severe bremsstrahlung problem.

The artificial belts decay rapidly at the belt/atmosphere interface (characteristic decay time about a week). At the middle of the artificial belt the decay time is several months. Years are required for the artificial belt to completely dissipate.

The electrons with energies greater than 0.5 Mev are influential in causing displacement damage (Section 4.2.2) and ionization effects (Section 4.2.1) in the electronics of a space system exposed to the radiation field for more than a few months after a burst.

4.1.1.6 Thermal Pulse

The high intensity light pulse is of little consequence from a pulsed source appreciably higher than 10 nmi. Accordingly, it can be considered as having negligible effect on the space tug. If the pulsed source occurred in the high density portion of the atmosphere, the thermal pulse which covers the entire UV, visible and IR spectra could seriously overload a high gain optical system on the tug.

4.1.1.7 Overpressure

The pressure has no effect on the tug at altitudes less than 10 nmi. insofar as nuclear pulsed radiation encounters are concerned.

4.1.2 Natural Space Radiation Environment

The mission profile of the space tug exposes the system to three sources of natural space radiation: galactic and extra-galactic cosmic particles, solar cosmic particles and flares, and the earth's natural radiation belt. This section describes aspects of these radiation fields pertinent to the space tug astrionic design.

4.1.2.1 Solar Cosmic Particles and Flares

The normal solar cosmic flux consists primarily of positively charged protons with average energies of approximately 100 mev per particle (Reference P-7). However, this proton flux intensity is relatively small and is additionally decreased in low earth orbit by the deflection of the protons by the earth's magnetic field. The most stringent environment that the space tug will encounter will occur during the synchronous orbit and RNS earth/moon missions and lunar orbit operations. Experience gained by the successes of Syncom II and Syncom III communications satellites which operated for years in synchronous orbit and by the success of the Apollo lunar landing missions has demonstrated that normal solar cosmic particles do not present a design problem to the space tug astrionics.

However, with increased proton energies and intensities which are characteristic of solar flare activity, both ionization and displacement damage can result. Shielding the space tug astrionics for protection against solar flares could impose a heavy cost and weight penalty. But, since:

- The geomagnetic field surrounding the earth provides some protection against solar flares,
- High energy solar flare events occur only on the average of approximately 9-10 times per year,
- There is some predictability for solar flare events, and
- There is lag time of several hours between optical detection of solar flares on earth and the ensuing significant radiation increase,

the space tug missions could be planned or the mission could be terminated to circumvent operating the space tug during a high intensity solar flare.

4.1.2.2 Natural Radiation Belt (Van Allen)

The Van Allen radiation belt consists mainly of energetic protons and electrons in a donut-shaped geometry surrounding the earth beginning at an altitude of about 200-500 nm and extending out to about 35,000-40,000 miles. Maximum proton and electron energy densities occur in the inner and outer regions of the belt, respectively; hence, the concept by some of the existence of more than one belt. The belt does not extend beyond earth latitudes of about 75° north or south and is thus open at the earth's polar regions (Reference P-7). The energy spectra of both species of radiation vary widely, but permit the consideration of shielding as a hardening approach.

Electronic devices exposed to this environment will suffer both ionization and displacement damage. Unlike the pulsed radiation or solar flare sources, trapped natural radiation induces damage effects cumulative over lengthy periods of exposure.

Since the low earth orbits of the space tug are below any significant portion of the natural radiation belt, the only space tug missions of concern are the ones which traverse through the belt and the synchronous orbit mission. Synchronous orbit altitude is in the outer, lower energy electron radiation region of the belt. The following analysis is based on the synchronous orbit mission and will thus reflect the most stringent radiation exposure criteria.

Table 4-1 describes when the more sensitive transistors begin to show radiation damage which takes the form of reduced gain characteristics.

Table 4-1.

*Type of Particle	Energy	Damage Exposure To Transistors
Protons	>20 mev	>10^{11} protons/cm^2
Electrons	5 mev	>10^{13} electrons/cm^2

*Damage to transistors decreases with decreasing electron energy. Damage increases, up to a point, on decreasing proton energy.

The maximum radiation exposure to the space tug outer skin during the synchronous orbit mission is as follows (based on 24 hours at synchronous orbit altitude):

Protons:

>30 mev: $(10^3 p/cm^2\text{-sec})\ (3.6 \times 10^3\ sec/hr)\ (4\ hr) = 1.4 \times 10^7 p/cm^2$
$0.1 - 5$ mev: $(10^8 p/cm^2\text{-sec})\ (3.6 \times 10^3\ sec/hr)\ (30\ hr) = 1.08 \times 10^{13} p/cm^2$

Electrons:

>1.6 mev: $(10^3 e/cm^2\text{-sec})\ (3.6 \times 10^3\ sec/hr)\ (32\ hr) = 1.15 \times 10^8 e/cm^2$
>0.04 mev: $(2.5 \times 10^8 e/cm^2\text{-sec})\ (3.6 \times 10^3\ sec/hr)\ (32\ hr) = 2.9 \times 10^{13} e/cm^2$

Liberal estimates were made on particle flux densities. Times for space tug exposure to various energy particles vary due to variation in particle flux densities with altitude.

Table 4-2 is a summary of total particle exposure.

As can be seen from comparing Tables 4-1 and 4-2, with the possible exception of relatively low energy protons, the radiation exposure to the space tug is insufficient to cause damage to space tug astrionics without taking credit for any shielding whatsoever. The range of 5 mev protons in aluminum (the space tug skin will likely consist of an aluminum alloy) is on the order of 0.02 cm. Less energetic protons would be less penetrating. The space tug skin will be thicker than this; thus the tug skin alone is capable of stopping protons of 5 mev energy and below.

Table 4-2.

Type of Particle	Energy	Total Exposure To Outer Skin
Protons	>30 mev	10^7 protons/cm^2
Protons	0.1–5 mev	10^{13} protons/cm^2
Electrons (Beta)	>1.6 mev	10^8 electrons/cm^2
Electrons (Beta)	>0.04 mev	10^{13} electrons/cm^2

In addition to the factors of safety mentioned above, such as a liberal computation for total space tug exposure and the use of the space tug skin as shielding material, the following items would act as shielding material to further reduce space tug astrionic component internals from radiation exposure:

- Cold plate panels and coolant
- Component mounting brackets
- Component enclosures
- Component enclosure environments (i.e., some components may be pressurized)
- Component internals will have a self-shielding effect upon electronic devices located farther toward the center of the component
- The space tug propulsion module and crew/cargo module will completely shield the top and bottom of the astrionic module
- Thermal insulation, louvers, radiators, etc.

Giving additional confidence in the operability of the space tug during the synchronous orbit mission is the fact that synchronous satellites (Syncom II and III) have been placed into synchronous orbit and have operated there for years.

4.1.2.3 Galactic or Extra-Galactic Cosmic Particles

These particle fluxes are comprised primarily of positively charged particles with extremely high energies. The estimated average energy is approximately 10^4 mev, thus making shielding impractical (Reference P-7). The intensities of these fluxes are very low, however, and therefore the ionization and displacement damage associated with them is negligible relative to the consequences of exposure to solar flares or the natural trapped radiation.

4.1.3 Nuclear Reactor Radiation

The space tug will be interfacing with NERVA engines (in the capacity of flying the RNS mission and when acting as a utility vehicle for servicing and maintaining RNS vehicles) and space base nuclear power reactors. Thus, the radiation exposure to the space tug from these sources must be considered as a contributor to the total radiation environment.

The space tug astrionic module in the vicinity of an operating reactor will be primarily exposed to highly penetrating gamma rays and neutrons. They will induce both ionization and displacement damage. For the most part, however, the problems posed by the reactor are similar to those of pulsed radiation with the exception of exposure time and directionality.

The most stringent radiation environment from nuclear reactors that the space tug will encounter will be in flying the RNS mission. Current NERVA/RNS design concepts indicate that at the location above the RNS propellant tank where the majority of the astrionics will be located, the radiation environment will be as follows (Reference P-8):

Gamma Rate – 1 rad/sec (3.6×10^3 rads/hr)

Gamma Dose – 3.6×10^4 rads

Neutron – Negligible

These levels are well within the state-of-the-art electronic technology and thus will impose no special design problems to the tug astrionics. Optical devices on the tug, such as landmark trackers, star trackers, etc., face outboard from the vehicle and thus should receive minimum radiation impingement on their optical face.

When the space tug is in proximity to an RNS or space base nuclear power reactor, appropriate operational constraints will have to be observed to keep the tug in positions of low radiation environments. This will have no significant impact on the tug, however, and will be simply normal safety operating procedures.

4.2 RADIATION EFFECTS IN COMPONENTS AND MATERIALS

The effects of radiation on electronic components and materials can be separated into four general classes: ionization, displacement, thermal, and mechanical. These classes were used in Section 4.1 to designate the principal effects of various species of radiation which would be encountered by a space tug. This section describes the damage mechanisms of the four classes of effects and identifies those devices and materials most susceptible to the space tug radiation environment. Approximate failure levels will be presented in cases where they have been adequately determined.

The ionization and displacement damage radiation levels which a space tug astrionic module can be designed to tolerate are given as a function of technology limitations in Table 4-3. The indicated radiation levels are presumed to be the combined environments discussed in Section 4.1 and which would be consistent with space tug mission profiles.

Table 4-3. Technology Limits Versus Radiation Levels

Combined Radiation Sources

Technology Limits	Ionization Effects: Dose rad (si)	Ionization Effects: Dose Rate Rads (Si)/Sec a	Ionization Effects: Dose Rate Rads (Si)/Sec b	Displacement Effects Fluence 1 Mev Silicon Equivalent Neutrons/cm^2
State-of-the Art Technology	10^6	10^{11}	3×10^8	10^{14}
Pushing Current Technology	2×10^6	5×10^{11}	10^9	3×10^{14}
Probable 1973 Technology	3×10^6	10^{12}	5×10^9	3×10^{14}

a) Circumvented System (Reference Section 4.3.2)
b) Uncircumvented System

The following subsections are concerned mainly with radiation effects on semiconductor devices. Optical equipment, such as star trackers, laser radars, etc., exhibit optical face discoloration which degrades the sensitivity of the device. However, radiation resistant glasses are available to minimize this effect. Some glass is purposefully designed to exhibit high electrical conductivity so that, when a small electrical current is passed through it, opaqueness is annealed out.

4.2.1 Ionization Effects

Most types of radiation interact, to a greater or lesser degree, with matter in such a way as to create charged particles in the absorbing medium which can respond to electric fields or diffuse. Semiconductor devices are quite sensitive to ionizing radiation and are of principal concern here. Dielectrics are somewhat less sensitive but are also a problem in sufficiently intense radiation fluxes. It is convenient to categorize ionization effects as either transient or permanent.

4.2.1.1 Transient Effects

PN junction semiconductor devices will produce a transient photocurrent (Ipp) which can cause circuit upset. In the space tug the most susceptible devices would be power transistors, power diodes, junction field effect transistors (JFET), junction isolated integrated circuits (particularly the linear types), silicon controlled rectifiers (SCR), and infra-red (IR) detectors. All these devices can be expected to suffer malfunction in their typical applications at dose rates less than 10^9 rad(Si)/sec unless appropriate hardening techniques are employed.

A critical nonsemiconductor device which will malfunction (depending on circuit tolerances) at relatively low dose rates is the quartz crystal. Since it is frequently employed as a clock oscillator, it would be expected to operate properly even in a circumvented system.

Capacitors will produce a transient leakage current or voltage reduction under the influence of an ionizing radiation pulse. This effect does not begin to become serious in most applications until dose rates exceed 10^9 rad(Si)/sec.

4.2.1.2 Permanent Effects

The devices most sensitive to permanent ionizing doses are the members of the insulated gate FET (IGFET) family. At pulsed radiation doses of 10^4 to 10^5 rad(Si) the threshold voltage begins to degrade rapidly due to a space charge buildup in the oxide layer. Because of long-term annealing, the damage from pulsed radiation can be considered permanent in a digital system. On the other hand, the same dose delivered in a slow, cumulative fashion, as with some types of space radiation and with radiation from NERVA, permits the IGFET to survive much higher doses of these types of radiation. At doses of a megarad (Si) the surface effects begin to measurably reduce the forward current transfer ratio of ordinary transistors. This is usually negligible compared to the same effect by neutrons. However, in a low neutron fluence environment, the effect could become significant.

4.2.2 Displacement Effects

The displacement of atoms and disordering of semiconductor crystal lattices by particulate radiation influences the electrical parameters of devices. The most profoundly affected parameters are forward current transfer ratio (Beta) and Vce (Sat) of power transistors. The only device which suffers from displacement damage at fluences of less than 10^{11} to 10^{12} neutrons/cm^2 is the unijunction transistor. Beta begins to degrade between 10^{12} and 10^{13}n/cm^2 for most silicon transistors. Fanout of integrated circuits begins to degrade at about 10^{13}n/cm^2. Vce(Sat) of high BVceo (greater than 80 volts) begins to increase rapidly at fluences between 5 x 10^{13} and 10^{14}n/cm^2. The forward voltage of power diodes increases at the latter fluence levels.

All of the above effects are permanent with the exception of the degraded Beta phenomenon. It is quasi-transient in that the Beta decreases to a minimum value at the beginning of the neutron pulse and within milliseconds increases to a relatively permanent value less than the pre-irradiation Beta. This Beta annealing is a problem when operation during high level neutron pulses is required.

It should be noted that electrons, protons, and secondary neutrons from the natural space radiation environment create the same damage but in a slow cumulative fashion and not as effectively as the pulsed radiation neutrons. However, if neutron fluences are low, on the order of 10^{11} to 10^{12}n/cm^2, the natural space environment may produce the most significant damage.

4.2.3 Thermal Effects

The rapid deposition of large amounts of energy can raise the temperature of certain component materials, causing temporary malfunction or permanent damage. If the neutron fluence is much less than 10^{14}n/cm^2, neutron heating will be negligible.

It should also be pointed out that the Vce(Sat) of power transistors is increased by a neutron induced increase in collector bulk resistance. The irradiated transistor thus operates at a higher temperature and power level. Again, this effect is negligible for neutron fluences much less than 10^{14} neutrons/cm^2.

The implications of all of the above are that thermal effects will not be significant unless the total single pulse dose approaches a megarad (Si).

4.2.4 Mechanical Effects

The only mechanical damage mechanism of concern for a manned system is the deterioration of certain polymers exposed to ionizing doses in excess of a megarad (C). Specifically, certain types of teflon will suffer chain scission and consequent softening of the material. Small amounts of flourine gas may evolve creating a potentially corrosive atmosphere in the electronic subsystem. The latter effect would require several megarads (C) before becoming a matter for concern.

4.2.5 NEMP Component and Circuit Radiation Effects

The definition of upset, as used here, is a circuit response which causes some electrical system or subsystem to malfunction which may be transient or permanent (Reference P-6). The analysis of the upset problem is usually straight-forward because the function of the circuit is well defined. The examination of the response of a bi-stable circuit used to store information serves as a good example. A transient through the power supply or a common wire may cause switching and loss of information. This is a permanent upset. A transient upset is one which produces an excursion and returns to the original operating condition.

On the other hand, burnout, as used here, refers to the permanent change in the characteristics of a device (Reference P-6). Burnout is a problem which can be examined analytically, using digital circuit analysis codes with an assist from experimental data obtained from actual tests on devices. To a large extent, burnout results from energy deposition in a component, for example, due either to excessively large reverse voltages or forward currents. Each class of components (transistors, diodes, etc.) has its own peculiar burnout mechanisms (Reference P-6).

Typical results with a 2N2222 transistor indicates that junction burnout occurs at an energy deposition level of approximately 100 microjoules with a 100 nanosecond pulse width (References P-11 and P-12). Typically, when a large reverse bias is applied to the base emitter junction from a low impedance (10 to 30 ohm) source, a current of hundreds of milliamperes will flow through the junction. Burnout can occur from pulsed sources greater than 50 miles distant when the junction areas are 10^{-3} square centimeters or smaller. A great deal of protection is required to prevent junction burnout. The energy is conducted into the circuits and components normally via long interconnecting cables. However, other mechanisms are also prevalent, such as inductive and capacitive coupling from currents racing up and down the vehicle skin. Also, antenna type coupling into components and circuits occurs.

Thus, in assessing the NEMP vulnerability of a system, one should realize that failure of the system to function in accordance with its design can occur not only on a subsystem level but on a component and circuit level as well (Reference P-10). It is desirable to establish a means of predicting the failure level of system components due to high amplitude voltage pulses, especially in solid state electronic devices such as diodes and various types of transistors. The next step is to identify the susceptible components. Components or subsystems subject to operational upset include the following:

- Low power or high speed digital processing systems,
- Memory units, especially for digital processing such as core memories, drum storage, and buffers via wiring,
- Control systems for in-flight guidance,
- Protection or control systems for the distribution of power,
- Subsystems employing long integration or recycling times for synchronization acquisition or signal processing, such as gyros.

Components subject to functional burnout include the following:

- Active electronic devices such as semiconductor devices, especially high frequency transistors, integrated circuits, and microwave diodes,
- Passive electrical and electronic components, especially low power or low voltage or precision type components,
- Semiconductor diodes or silicon controlled rectifiers, especially in power supplies connected to long cables,
- Meters, indicators, or relays, especially those employed to control power, flight, or guidance,
- Insulated RF and power cables, especially those running near maximum rating, and
- Components connected in systems containing large amounts of stored electrical energy.

4.3 RADIATION HARDENING

The purpose of radiation hardening is to provide a system or subsystem capable of performing required functions during and after exposure to a radiation environment. Before hardening can begin, the environment must be defined in sufficient detail to determine radiation effects of component and circuit technologies that could be used to implement the system. A trade-off effort considering various system configurations and modes of operation, various component and circuit technologies and various hardening techniques should then be conducted to select the hardening approach to be used. Some combination of the hardening approaches discussed in this section will usually provide the most effective (cost, weight, performance) approach.

4.3.1 Intrinsic Hardening

The intrinsic hardness level has been defined as that level of radiation at which no effect on circuit operation occurs. For prompt ionizing pulses this level is the transient malfunction level of a circuit. Such transient malfunctions may effect system operation in any of the following ways:

- Generate significant noise signals,
- Cause transient errors which may degrade system operation or result in system failure or mission abortion, or
- Degrade or burnout components which may cause system degradation or failure.

The NEMP generates voltage and current surges in antennas, cabling, and wiring which can also cause circuit malfunctions and component degradation or burnout.

The intrinsic hardness level for parameters of the radiation environment which cause permanent degradation of components and materials (neutrons, ionized dose, etc.) is that level at which circuit operation or component parameters are no longer within specification.

System hardening to levels greater than the intrinsic hardness level of transient effects can be achieved. Intrinsic hardness levels for permanent effects usually limit system hardware even with component, circuit or unit redundancy since most permanent degradation occurs with or without power applied.

4.3.2 Circumvention

Circumvention has been defined for a digital computer as a combination of hardware and software concepts which enables a computer to suffer an unprogrammed temporary interruption as a result of transient malfunctions without appreciably affecting the solution of the real-time problem. This approach is not limited to a digital computer, however, and can be used to harden various subsystems. For a system utilizing a digital computer, circumvention is often the most cost effective hardening approach especially at high radiation levels where intrinsic hardening becomes very difficult.

Circumvention requires a radiation detector and the hardware and software to reinitialize or restart. The radiation detector issues a clamp signal whenever the radiation exceeds a critical or circumvention level and maintains the clamp for a fixed period of time or until radiation has dropped below the critical level and circuits have recovered. Elapsed time can be obtained from a resistor-capacitor timer as part of the detector circuitry or from a special hardened timer if greater accuracy is required. Some form of hard memory (drum, plated wire, tape, read only) that will retain the data required for restart and past data to update after restart is required.

Typically it takes up to a microsecond or more to detect a critical level and generate a clamp signal and much longer to turn off power supplies. Errors and any component or circuit degradation or burnout that could occur during this time must be considered in any hardening effort.

Multiple pulsed radiation time spacing must also be considered. If a second pulsed radiation occurred during restart, and the systems were not designed for this case, significant system errors or even system failure could occur.

4.3.3 Shielding

Shielding in space systems is one of the most effective hardening techniques against all types of radiation with the exception of fast neutrons and gamma rays. For example, the thickness of shielding required to attenuate the dose from a fission gamma spectrum flux by an order of magnitude would be about 6 to 8 cm of lead or 3 to 5 cm of tantalum.

Unfortunately, shielding is also one of the most complicated methods, involving considerations of the total system architecture. Structural frames, plumbing, fuel tanks, or any matter intervening between the electronics and the external radiation environment modifies and transforms that environment. For example, the structural frame and component equipment casing will in general shield out a major portion of the particulate radiation fluxes of the natural space environment. In doing so, however, another type of radiation, bremsstrahlung, is created, which is a high energy (10-100 Kev) X-ray. This accounts for the large ionizing doses in electronics although relatively little of the original flux penetrates through to the electronic subsystem.

It should be recognized that it is not always necessary to shield an entire subsystem. A combination of local shielding inside equipment cases and other hardening techniques is frequently the most cost effective approach in a system where weight reduction is of high priority. It is, as a rule, much less expensive to design local shielding, and it is much less expensive than the design of special radiation hardened devices which might otherwise be required.

4.3.4 Specific Hardening Techniques

One or more parameters of the radiation environment contribute to each basic category that must be considered in a hardening effort. The following categories are discussed in this section:

- Ionized dose rate
- Ionized dose
- Displacement effects
- NEMP
- Thermal and mechanical effects.

4.3.4.1 Ionizing Dose Rate

Ionization radiation generates photo currents (Ipp) in semiconductor devices, increases the conductivity of capacitor dielectrics and causes charge scattering and secondary emission in components, materials and circuit assemblies. These effects cause circuit transients and at high levels can cause device degradation or burnout. All of the following approaches can be used to harden to ionizing dose rate effects.

4.3.4.1.1 Component Selection. The large transient current that flows across a reverse-biased semiconductor junction is one of the most significant effects of the ionizing dose rate. High frequency, small geometry semiconductors operated at low voltages will minimize these effects. Minimum Ipp will be generated in a transistor with the following characteristics:

- Small junction area
- Small collector base transition region
- Epitaxial collector with a minimum thickness and resistivity
- Gold doping to minimize minority carrier lifetime.

Typical values for logic transistors designed for minimum Ipp range from 20 to 80 microamperes at an ionizing dose rate of 10^9 rads (Si)/sec. Approximately the same current levels for charge scattering and secondary emission are typical for this dose rate. Ipp in the milliampere and ampere range are typical for linear transistors and medium and high power devices at 10^9 rads (Si)/sec.

Photosensing and photo-detecting devices are very sensitive to ionizing radiation with photocurrents equaling or exceeding normal operating currents at rates as low as 10^5 to 10^6 rads (Si)/sec.

Component selection is always constrained by functional requirements, such as power rating, breakdown voltage, gain, etc. that the circuit designer must consider in any design effort.

4.3.4.1.2 Circuit Design. Most unhardened linear type circuitry will malfunction at 10^6 to 10^8 rads (Si)/sec while unhardened logic and switching type circuitry will malfunction at 10^7 to 10^9 rads (Si)/sec. If radiation effects are included as criteria in the initial design effort, circuitry can be designed with an intrinsic hardness of 10^8 to 10^{10} rads (Si)/sec and with no burnout at 10^{11} to 10^{12} rads (Si)/sec.

In addition, for many circuit applications functional requirements can be achieved, even with short duration transients, if these effects are considered in the initial design phase.

Some specific circuit hardening techniques are:

- Photocurrent cancellation or compensation
- Low impedance circuitry
- Differential balancing
- Current limiting to prevent burnout at high dose rates

Additional techniques are presented in the TREE HANDBOOK (Reference P-15) and numerous circuit and system hardening study contracts. All of these techniques were used in the Advance Hardened Guidance Computer effort (Reference P-16).

4.3.4.1.3 Technology Selection. Microelectronics selected from available commercial units will be used extensively in space tug astrionics. Some radiation hardened circuits are becoming available, but selection of technologies and specific circuit types and configurations from non-hardened units can also be used to harden electronic systems.

Dielectric isolation eliminates substrate photocurrents, thus increasing intrinsic hardness levels, eliminates most potential latchup problems and minimizes burnout problems at high dose rates. High speed logic circuits (high currents, low resistance values) are harder than low power (low current, high resistance) circuits.

A gold doped dielectric-isolated unit using thin film resistors is one of the hardest technologies. However, even with this technology it is difficult to intrinsically harden linear circuits (sense amplifiers, differential amplifiers, operational amplifiers) to levels greater than 10^8 to 10^9 rads (Si)/sec.

4.3.4.2 Total Ionized Dose

The two most significant effects of the total ionized dose are degradation of metal oxide silicon FET (MOSFET) characteristics and capacitor discharge. Selected state-of-the-art MOSFETs should be usable at doses between 10^4 and 10^5 rads (Si) with production controls and screening. Some of the technologies now being developed offer promise of satisfactory operation at or above 10^5 rads (Si). Bipolar transistors and JFETs can be used at doses of 10^6 to 10^7 rads (Si)'

Capacitor ionization effects can be modeled with a current source connected across the capacitor terminals in a direction to discharge the capacitor. This current is a function of both dose and dose rate as well as capacitor voltage and the dielectric material. Typical discharge ratios (voltage final/voltage initial) for common capacitor types will vary from 90 to 40 percent for doses ranging from 10^4 to 10^6 rads. If the capacitor is used in a circuit where some charging current is available, the voltage loss will be less significant. Selection of capacitor dielectric for critical applications can minimize this effect.

4.3.4.3 Displacement Damage

Displacement damage can result from neutrons, electrons or protons and is usually treated in terms of 1 mev equivalent neutron damage. Gain degradation of bipolar transistors and changes in other semiconductor characteristics occur at 10^{11} to 10^{12}n/cm^2 and become severe at 10^{13} to 10^{14}n/cm^2. Transistor gain degradation is a major problem. Significant changes in other transistor characteristics become a problem only at levels where severe gain degradation has occurred.

4.3.4.4 NEMP Hardening Techniques

In order to harden the space tug to the NEMP from a pulsed radiation, it is necessary to obtain an estimate of the electromagnetic field environment in the neighborhood of the equipment, the induced currents flowing into the equipment on the interconnecting cables, and the NEMP induced into the system by antenna coupling mechanisms (Reference P-6).

The shielding effectiveness of the metal structures surrounding the electronic equipment must be ascertained. This can be accomplished in two ways: analytically and/or experimentally. Using either of these approaches, the shielding effectiveness of the space tug can be ascertained. Typically, the shielding effectiveness of geometries similar to the space tug give shielding attenuations ranging from 20 to 70 db depending on pulsed source size, apertures and discontinuities, the frequency of the incident radiation, skin thickness, material properties, and other parameters. The incident fields are reduced by one order of magnitude for each 20 db attenuation.

The primary NEMP system vulnerability is to high voltage and current surges on electronic components and circuits (Reference P-6). These voltage and current surges can be several thousands of volts and hundreds of kiloamperes, respectively (Reference P-12). Such induced voltages and currents can burn out associated electronics, puncture cable insulation, degrade critical electrical components, blow out fuses and circuit breakers, disrupt critical signals in cables, and cause temporary or permanent damage to all connected electrical equipment not protected from current and/or voltage surges produced by the NEMP. This type of interference entering via the cables is normally designated as the conducted NEMP and is known to have the most detrimental effect on electronics such as the proposed tug astrionics.

Hardening techniques for conducted NEMP are: proper cable arrangements (Figure 4-2), good grounding and bonding techniques, cable shielding, cable surge arresters and protectors (Figures 4-3 and 4-4), and cable minimization.

Since the space tug contains both directional and omnidirectional antennas, antenna coupling of energy into system cables and electronics must be considered (Reference P-6).

Miscellaneous NEMP hardening techniques which have been employed on operational systems are:

- Good RFI design and practices were employed.
- The system was tested at threat levels.
- An NEMP control plan was written.
- Ground loops were eliminated by transformer decoupling, differential circuitry, and shielded twisted pairs.
- Power lines were single-point grounded, paired, and twisted.
- Signal lines were transformer decoupled, shielded and twisted balanced pairs.
- The analog lines were single-point grounded.
- Cable shields were grounded at one end to reduce capacitive coupling and IR drops in the sheath.
- Circumvention was employed on the power lines.
- Design changes included circumvention, cable shielding, and a TREE wiring arrangement.

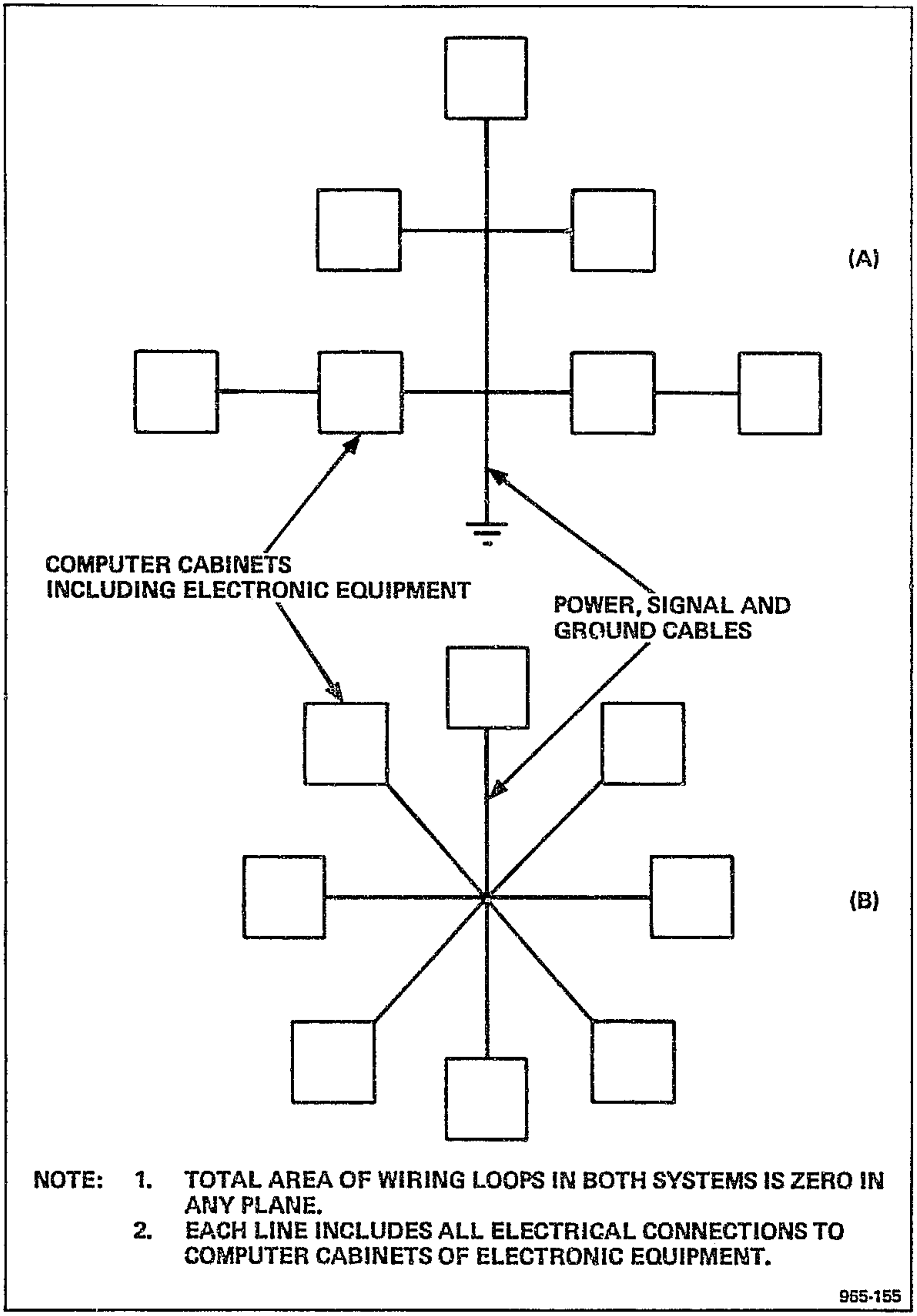

Figure 4-2. (A) TREE, (B) RADIAL Wiring Systems for Reducing NEMP Susceptibility

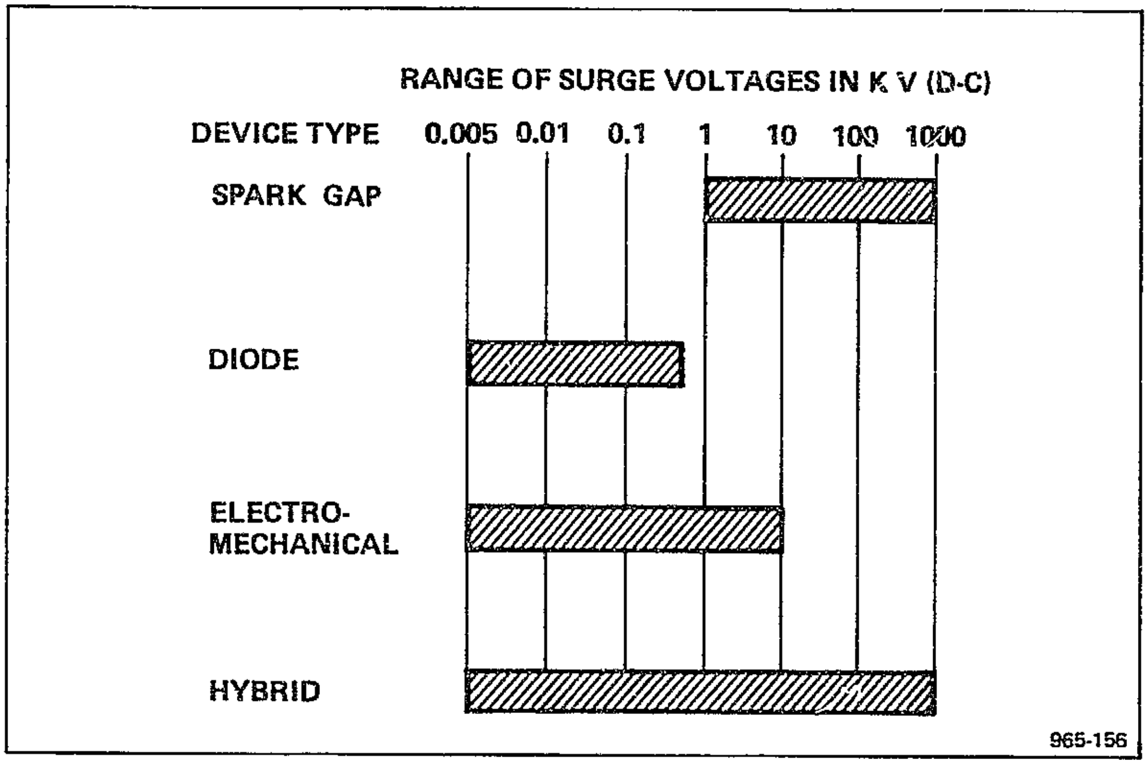

Figure 4-3. Operating Ranges of Generic Protection Devices

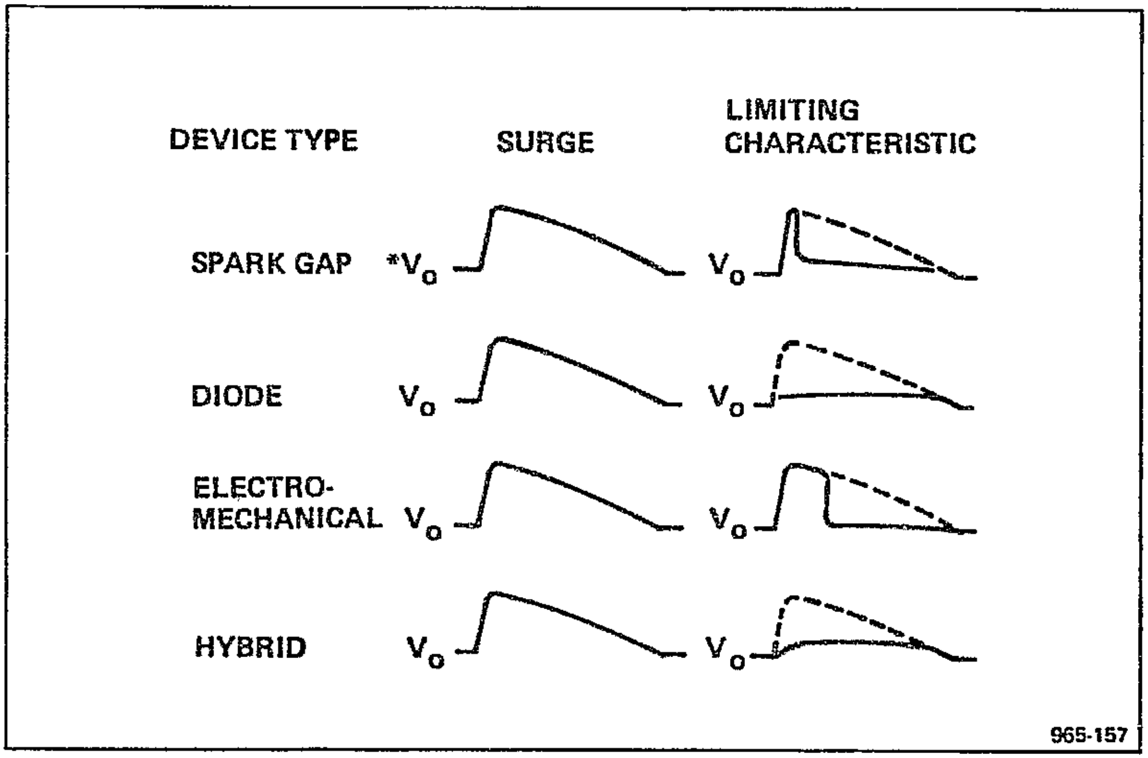

Figure 4-4. Limiting Characteristics of Generic Protective Devices

- Digital computer circuitry changes include: single-point grounding, signals fed by pulse transformers to shielded twisted pairs, logic circuit clamps on umbilical disconnects, minimized feedback by low impedance drivers for pulsed transformers, single end grounded cable shields to circuit ground, capacitor decoupled power lines in all computer subunits, and twisted power leads to batteries.

- Initial tests were performed by pulsing currents on the vehicle skin.

- System analyses include: single-point grounding, all power lines grouped and twisted, signal wires transformer decoupled and shielded, single end cable shield ground, input/output signals on balanced pairs and transformer decoupled, clamps on all open circuits to the computer, and separate ground systems for the computer.

- Re-design and re-testing, fixes, and final tests were performed on the system before final acceptance.

- Other hardening techniques employed include: shielding, surge protection, overdesign, the isolation matrix technique, orientation, grouping, cable routing, energy comparisons, elimination of ground loops, RFI shielding protection, and fiber optics.

4.3.4.5 Thermal and Mechanical Effects

Thermal and mechanical effects were covered in sections 4.2.3 and 4.2.4, respectively. They are a problem only at high radiation levels when little or no effective shield between components and the incident radiation is provided. Shielding and component and technology selections are used to harden these effects.

4.4 SYSTEM HARDENING PROGRAM

A typical system radiation hardening program is outlined in Table 4-4. The effort required for each task is dependent upon the type of system to be hardened, the radiation levels and system hardening approach selected.

4.4.1 Design and Prototype Development

The program outlined in Table 4-4 is presented to indicate major points of a hardening effort. Such an effort must be implemented to be compatible with the overall system development and production plan.

4.4.1.1 Define Environment

Although the environment incident upon a vehicle is specified, the details required for hardening are usually not provided. The pulse shapes or time histories for the types of radiation causing transient effects (gamma, X-ray, NEMP) are needed to specify component effects and to harden circuits and systems to these effects. The time spacing of any multiple pulsed radiation environment is critical to any system hardening effort.

Table 4-4. System Hardening Program

DESIGN AND PROTOTYPE DEVELOPMENT
Define Environment
Select System Hardening Approach
Generate Radiation Effects Design Data
Piece Parts Radiation Effects (Testing) Design Guidelines
Hardened Design (Analysis and Testing)
Circuit Selection Circuit Design Shielding Cabling Circumvention Hardware and Software
Piece Part Radiation Qualification Testing
Unit and Subsystem Qualification Testing
System Testing
PRODUCTION
Piece Part Radiation Qualification Control
Vendor Controls Radiation Testing Screening
Unit and Subsystem Testing
System Testing

The radiation levels in component materials and at wire, cable and circuit locations internal to the vehicle and inside cases or cable shields must be known to specify radiation effects. For some radiation types (X-rays, electrons, NEMP) the vehicle structure, cable shields, and equipment cases significantly modify the radiation levels and, consequently, radiation effects. The energy spectrum and pulse shapes are also significant and must be considered.

4.4.1.2 Select System Hardening Approach

With the environment defined and some data available on component, material and circuit radiation effects, a trade-off study to determine the system hardening approach can be initiated. It is usually far more effective to consider radiation hardening as an additional requirement during initial system definition than to harden once components, circuitry, and functional units have been specified.

4.4.1.3 Generate Radiation Effects Design Data

This effort will provide the designer (electrical, mechanical, installation, system) with the detailed radiation effects data required to select and design circuits, assemble and package equipment and configure units into a hardened functional system. Some radiation testing of components and microcircuits is usually required.

Various radiation simulators or test facilities such as flash X-ray machines, linear accelerators, fast burst reactors, and NEMP sources are available for radiation testing. A list of facilities can be found in References P-6 and P-17 while standard procedures for testing at these facilities are presented in References P-6 and P-18.

4.4.1.4 Hardened Design

The hardened design effort required is dependent upon the radiation levels and the system approach selected. If circumvention is used, the additional hardware and software to implement this approach must be designed. Both analysis and radiation testing will be required to evaluate and verify design procedures. Automatic circuit analysis programs such as SCEPTRE (Reference P-19) and PANE (Reference P-13) can be used for hardened circuit design efforts. Computer analysis can also be used to determine shielding effectiveness and for NEMP hardening.

4.4.1.5 Piece Part Radiation Qualification

Semiconductor devices, microcircuits, and any other components or materials that will be significantly affected by the specified radiation environment and that are to be used in the production version of the system must be radiation qualified. The extent of this effort, which would become a part of the component assurance program, is dependent upon the radiation levels specified. Testing of significant sample lots at various radiation facilities is required for radiation qualification.

4.4.1.6 Unit and Subsystem Qualification Testing

Radiation testing is required whenever significant radiation levels are specified as part of the total environment. Usually standard production test equipment and procedures can be modified and adapted for radiation testing of units and subsystems.

Transient testing (ionizing pulse, NEMP) requires that units be operating and their performance evaluated during and immediately after radiation. Since test equipment will also be affected by the radiation, it must be shielded or connected remotely by cabling. Such testing is usually more difficult and expensive than normal production qualification testing.

Testing for displacement effects can usually be done on a pre-test/post-test basis, irradiating units with no power applied and evaluating performance before and after radiation with standard test equipment. This type of testing will degrade components and materials; therefore, units should not be used for other qualification tests after radiation. Refurbished units from other environmental tests (temperature, vibration) could be used effecting some cost savings.

4.4.1.7 System Testing

Most radiation testing can be done on a component, circuit or unit level. However, NEMP testing to verify system hardness is only valid for a complete vehicle or portion of a vehicle with all equipment interconnected in its final configurations and exposed in a NEMP simulator. Some NEMP testing of components, circuits, cables and units is useful in designing and evaluating probable system hardness but is not sufficient to verify NEMP system hardness.

4.4.2 Production

Production controls to assure that radiation hardness designed into the system is maintained throughout the production phase is required.

4.4.2.1 Piece Part Radiation Qualification Control

A control plan that insures that piece parts meet radiation specifications is necessary. Usually this can be limited to those components exhibiting significant effects (primarily semiconductors) at the specified radiation levels. Either a vendor program to supply qualified parts or a user quality control program can be used. For relatively low levels of radiation, process controls and selection based on critical parameters should prove adequate for most semiconductor devices and microcircuits. Periodic random lot sampling and radiation testing should be a part of any qualification control program.

The magnitude of such a program is very dependent upon radiation levels, the system hardening approach selected and the circuit technology used. For critical components or high radiation levels, non-destructive scanning tests or extensive sample lot testing will be necessary. For example, latchup can occur in junction isolated microcircuits at high ionizing dose rates (10^{10} to 10^{12} rads (Si)/sec) and screening tests may be required. State-of-the-art MOS devices will require extensive sample lot or screening tests for ionized dose effects at doses of 10^4 rads (Si) or greater.

4.4.2.2 Unit and Subsystem Testing

Testing for both transient effects and permanent degradation on a periodic basis using randomly selected units is required. Since transient testing can be non-destructive, each production unit could be checked if necessary. A comprehensive piece part quality control program can minimize the unit testing required.

If a circumvention system hardening approach is used, extensive testing of the circumvention hardware is necessary. Ionizing dose rate and NEMP detectors require testing to calibrate or set threshold levels and verify operation of the complete detector unit.

4.4.2.3 System Testing

NEMP system testing will be required if any configuration changes between the prototype and production system or other modifications to cables, circuits, or units that affect NEMP hardening have been made. A test of all units critical to circumvention and recovery connected in an operating configuration may also be required. If circumvention for both ionized dose rate and NEMP is required, testing in both environments is appropriate.

4.5 HARDENING PENALTIES

It is difficult to estimate the ramifications of hardening in terms of cost, weight, volume, power, etc., until a system hardening approach has been selected. In fact, these items are some of the things that are considered when performing a trade-off study to select a hardening approach. Some rough estimates of penalties and factors that must be considered are presented in this section.

4.5.1 Cost

Hardening costs based on Army, Navy and Air Force queries can be summarized as follows: The average percent increase ranges from 5 to 20 percent of the total system cost (including design, testing, production, etc.) for a hardened over a non-hardened system. It should be emphasized that this is only an average cost; thus, the actual costs for the space tug astrionics could be less or greater depending on the environment encountered, the astrionic design, and other factors. The number of systems to be built is not a factor since many of the hardening costs in design and development are fixed and not dependent on the number of production units.

All the items listed in the hardening plan (Section 4.4) will affect costs. Hardened design and the generation of the radiation effects data to support this effort can increase and even double the design costs over an unhardened system. It is estimated that piece part radiation qualification testing would cost $10 to $15 per part or $100 to $150 for a 10-part lot size for each type to be qualified from any one source. Unit and subsystem testing could cost from 20 to 100 percent more than a standard environmental test and an additional set of equipment may be required since some testing degrades components.

It is estimated that radiation qualified piece parts bought to specification would cost 15 to 20 percent more than similar parts with no radiation specification. Usually only semiconductors and a few special items must be procured to radiation specifications.

NEMP hardening can be considered as a "trial and error" approach. For example, present practice is to incorporate as many of the NEMP protective features as possible early in the design phase, then test and correct any problems. NEMP hardening costs are often a significant portion (1/3 to 1/2) of total hardening costs.

4.5.2 Weight

NEMP hardening and shielding for other types of radiation (X-ray, electrons) will usually contribute most to any weight increases. The weight penalty can be enormous if the protection problem is not approached until late in system design. The best way to minimize weight is to include hardening in the initial design. The cable weight is normally increased by about a factor of two for cable shielding; surge protectors increase the weight by a few ounces per protector. X-ray shielding, if required, is usually a few mils layer of tantalum or some other high density material. The weight is dependent upon the surface area of the equipment to be shielded. Circumvention hardware should increase weight by a small amount (a few pounds).

Hardening to severe radiation levels (10^{10} to 10^{12} rads (Si)/sec, 10^{13} to 10^{14}n/cm^2) will limit the choice of components and technologies and require special circuit and system designs, thus significantly increasing equipment weight. The percent increase is a difficult number to ascertain since it depends on the overall system size, the radiation environment, and other intangible factors.

4.5.3 Volume

The same factors increasing weight will also increase the volume.

4.5.4 Power

The component and technology selection and circuit techniques for hardening require low impedance, high frequency type circuitry operated at high current or power levels. Additional components for photocurrent compensation or increased gain and special memory technologies such as drum or plated wire may also be required. The power requirements for such a hardened system could be much higher than an unhardened system designed for minimum power. Power increases can only be determined when specific hardening requirements can be compared to an unhardened version of the system.

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APPENDIX Q

ASTRIONIC MODULE COST ESTIMATES WITH WORK BREAKDOWN STRUCTURE

IBM No. 69-K44-0006H
MSFC-DRL-008
LINE ITEM No. 268

TABLE OF CONTENTS

1.0 INTRODUCTION

1.1 OBJECTIVE

The primary objective of this appendix is to present preliminary recurring cost estimates for currently defined items of a work breakdown structure (WBS) developed for the space tug astrionic module. These cost estimates are presented in WBS format so that cost estimates for additional items addressed in future phases of this study can be added to the costs presented in this report. The WBS provides a consistent baseline for development and summation of cost data developed in this phase and future phases of space tug studies. A secondary objective is to provide a very preliminary look at total space tug astrionic module costs.

1.2 PREVIEW OF PRESENTATION

Section 2.0 below contains a tentative WBS outline for the space tug program in block diagram form and a more detailed WBS for the astrionic module in tabular form Section 3.0 presents preliminary estimates of recurring unit costs in WBS format for currently defined portions of three alternate configurations of the astrionic module. A rough estimate of the space tug astrionic module non-recurring cost is presented in Section 4.0.

2.0 WORK BREAKDOWN STRUCTURE

2.1 DERIVATION OF WBS

The work breakdown structure (WBS) proposed for use on space tug is based on the preliminary space shuttle program WBS presented in the "Astrionic System Optimization and Modular Astrionics for NASA Missions after 1974 – Monthly Progress Report for the period August 16 to September 18, 1969," MSFC No. III-6-602-107.

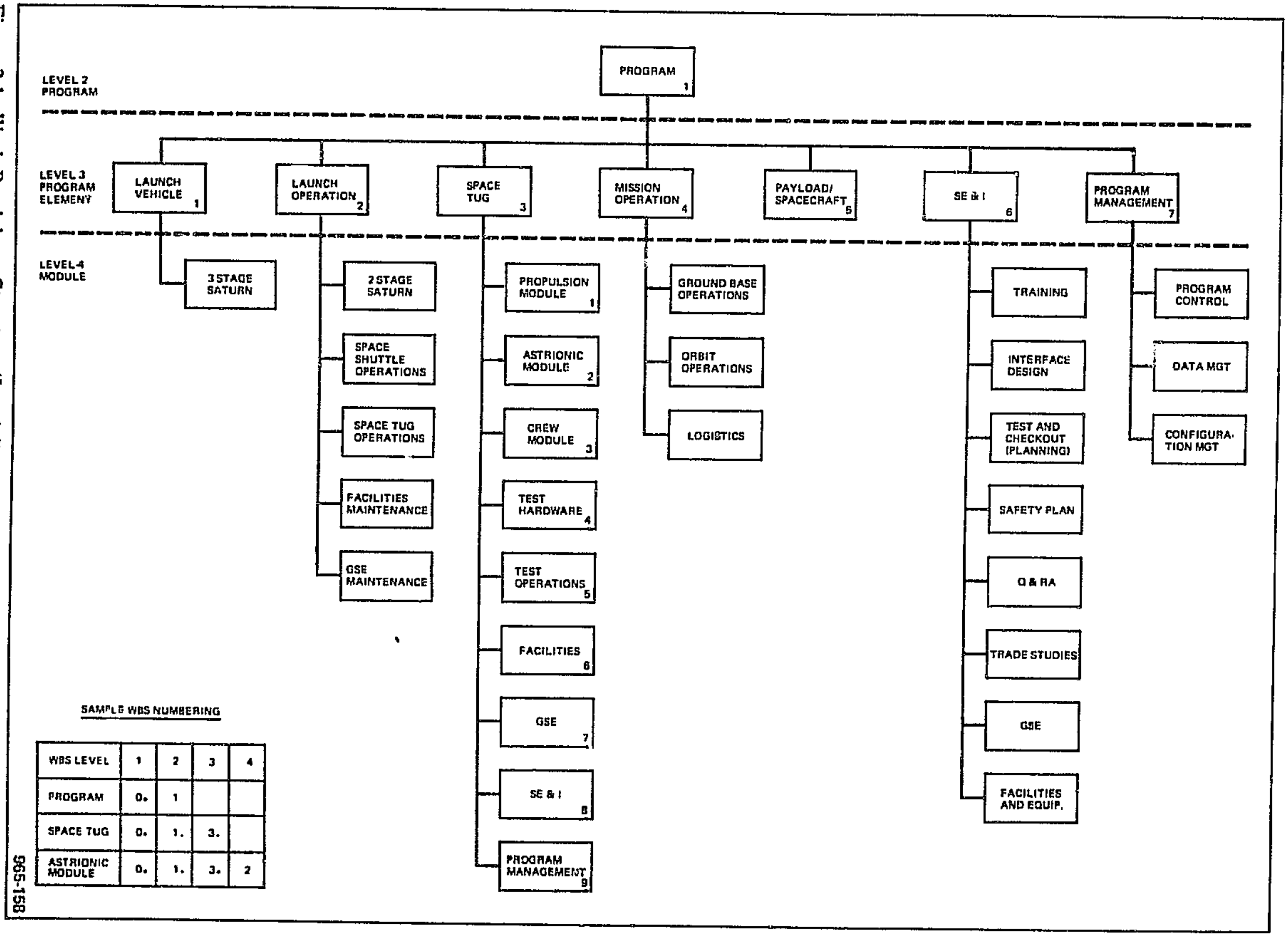
2.2 WBS BLOCK DIAGRAM

An outline of the proposed space tug WBS in block diagram form is presented in Figures 2-1, 2-2, and 2-3. Each chart contains an insert showing a sample of the WBS numbering system used. Numbers have been applied to only those blocks involved in the tiering of space tug astrionics costs.

2.3 WBS LISTING

Table 2-1 is a listing of all identified items in the WBS which are involved in the tiering of costs of the astrionic module and its identified major systems (level 5), subsystems (level 6) and assemblies (level 7). In several areas under the astrionic module (level 4), the WBS listing stops at level 5 or 6. This indicates that the lower levels are not presently defined. Components (level 8) have not been identified at this time and thus are not listed.

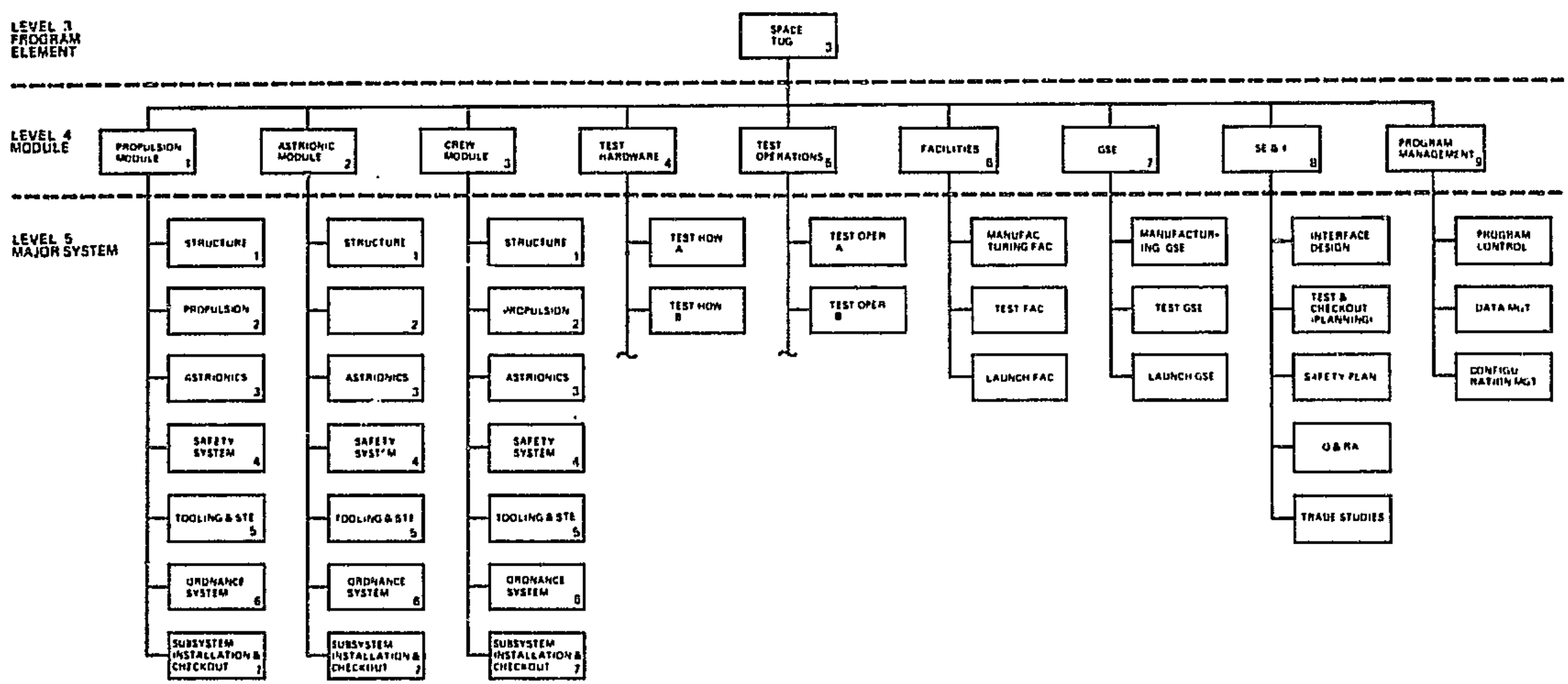
The "Technical Characteristics Data Form" taken from the documents referenced under paragraph 2.1 above has been used as the format for the WBS listing (Table 2-1). This form is designed for the presentation of technical, physical and mission characteristics of the WBS items. The form is used in this report as a WBS listing only, and thus the columns intended for listing of technical characteristics are blank. This form was chosen for use in this report for convenience and to provide a ready format for the listing of technical characteristics in future space tug astrionic system studies.



SAMPLE WBS NUMBERING

WBS LEVEL	1	2	3	4
PROGRAM	0.	1		
SPACE TUG	0.	1.	3.	
ASTRIONIC MODULE	0.	1.	3.	2

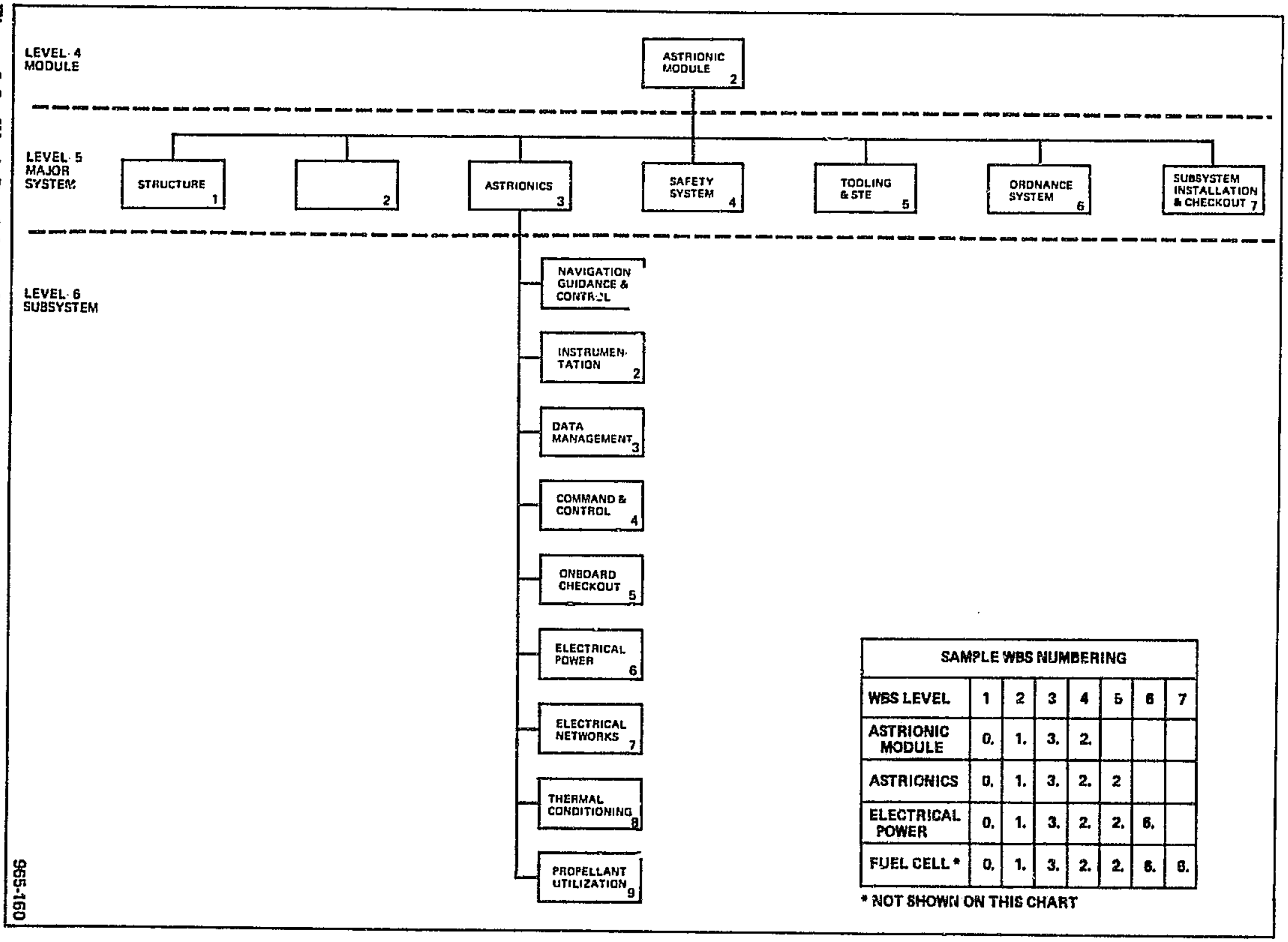
Figure 2-1. Work Breakdown Structure (Level 4)



SAMPLE WBS NUMBERING					
WBS LEVEL	1	2	3	4	5
SPACE TUG	0.	1.	3.		
ASTRIONIC MODULE	0.	1.	3.	2.	
ASTRIONICS	0.	1.	3.	2.	3

965-159

Figure 2-2. Work Breakdown Structure (Level 5)



SAMPLE WBS NUMBERING

WBS LEVEL	1	2	3	4	5	6	7
ASTRIONIC MODULE	0.	1.	3.	2.			
ASTRIONICS	0.	1.	3.	2.	2		
ELECTRICAL POWER	0.	1.	3.	2.	2.	6.	
FUEL CELL *	0.	1.	3.	2.	2.	6.	8.

* NOT SHOWN ON THIS CHART

Figure 2-3. Work Breakdown Structure (Level 6)

Table 2-1. Technical Characteristics Data Form

Identification Number	WBS Identification	Quantity or Value			Units of Measure	Characteristics	Notes
		WBS Level	Current	New			
0.1	Program	2					
0.1.1	Launch Vehicle	3					
0.1.2	Launch Operation	3					
0.1.3	Space Tug	3					
0.1.3.1	Propulsion Module	4					
0.1.3.2	Astrionic Module	4					
0.1.3.2.1	Structure	5					
0.1.3.2.2		5					
0.1.3.2.3	Astrionics	5					
0.1.3.2.3.1	Navigation, Guidance & Control	6					
0.1.3.2.3.1.1	IMU	7					
0.1.3.2.3.1.2	Laser Radar	7					
0.1.3.2.3.1.3	Star Tracker	7					
0.1.3.2.3.1.4	Landmark Tracker	7					
0.1.3.2.3.1.5	Horizon Sensor	7					
0.1.3.2.3.1.6	Landing Radar	7					
0.1.3.2.3.1.7	Rate Gyros	7					
0.1.3.2.3.1.8	Accelerometers	7					
0.1.3.2.3.2	Instrumentation	6					
0.1.3.2.3.3	Data Management	6					
0.1.3.2.3.3.1	CPU	7					
0.1.3.2.3.3.2	Bus Ctrl Unit	7					
0.1.3.2.3.3.3	Main Memory	7					
0.1.3.2.3.3.4	Mag Tape	7					
0.1.3.2.3.3.5	Display Memory	7					
0.1.3.2.3.3.6	C. Assign. Unit	7					
0.1.3.2.3.3.7	Aux. Mont. Comput.	7					
0.1.3.2.3.4	Command & Control	6					
0.1.3.2.3.4.1	USB Elect.	7					
0.1.3.2.3.4.2	USB Diplexer	7					
0.1.3.2.3.4.3	USB Antenna Switch	7					
0.1.3.2.3.4.4	USB Power Divider	7					
0.1.3.2.3.4.5	USB Omni Antennas	7					
0.1.3.2.3.4.6	USB Hi-Gain Ant. Ctrl	7					
0.1.3.2.3.4.7	USB Hi-Gain Ant.	7					
0.1.3.2.3.4.8	VHF Transceiver	7					
0.1.3.2.3.4.9	VHF Diplexer	7					
0.1.3.2.3.4.10	VHF Power Divider	7					
0.1.3.2.3.4.11	VHF Omni Ant.	7					
0.1.3.2.3.4.12	Cmd. Decode Elec.	7					
0.1.3.2.3.4.13	TV Camera	7					
0.1.3.2.3.4.14	TV Camera Ctrl	7					
0.1.3.2.3.4.15	VHF Command Rcvr	7					
0.1.3.2.3.5	On-Board Checkout	6					
0.1.3.2.3.6	Electrical Power	6					
0.1.3.2.3.6.1	Fuel Cell	7					

Table 2-1. Technical Characteristics Data Form (Continued)

Identification Number	WBS Identification	Quantity or Value			Units of Measure	Characteristics	Notes
		WBS Level	Current	New			
0.1.3.2.3.6.2	Reactant – H_2	7					
0.1.3.2.3.6.3	Reactant – O_2	7					
0.1.3.2.3.6.4	H_2 Tank	7					
0.1.3.2.3.6.5	O_2 Tank	7					
0.1.3.2.3.6.6	Battery	7					
0.1.3.2.3.6.7	DC Regulator	7					
0.1.3.2.3.6.8	Battery Charger	7					
0.1.3.2.3.7	Electrical Networks	6					
0.1.3.2.3.7.1	SIU	7					
0.1.3.2.3.7.2	Data Bus	7					
0.1.3.2.3.7.3	Monitoring Unit	7					
0.1.3.2.3.7.4	Aux Mon. Unit	7					
0.1.3.2.3.7.5	Power Distr.	7					
0.1.3.2.3.7.6	Aux Power Distr.	7					
0.1.3.2.3.7.7	Wire & Cables	7					
0.1.3.2.3.7.8	Junction Boxes	7					
0.1.3.2.3.7.9	Audio Subsystem	7					
0.1.3.2.3.8	Thermal Conditioning	6					
0.1.3.2.3.8.1	Coolant Pump	7					
0.1.3.2.3.8.2	Service Heat Exchanger	7					
0.1.3.2.3.8.3	Coolant Accum.	7					
0.1.3.2.3.8.4	Radiator	7					
0.1.3.2.3.8.5	Coolant Fluid	7					
0.1.3.2.3.8.6	Louvers	7					
0.1.3.2.3.8.7	Comp. Mtg. Panel	7					
0.1.3.2.3.8.8	Misc. Plumbing	7					
0.1.3.2.3.8.9	Multilayer Insulation	7					
0.1.3.2.3.9	Propellant Utilization	6					
0.1.3.2.4	Safety System	5					
0.1.3.2.5	Tooling & STE	5					
0.1.3.2.6	Ordnance System	5					
0.1.3.2.7	Subsystem Installation & Checkout	5					
0.1.3.3	Crew Module	4					
0.1.3.4	Test Hardware	4					
0.1.3.5	Test Operations	4					
0.1.3.6	Facilities	4					
0.1.3.7	GSE	4					
0.1.3.8	SE&I	4					
0.1.3.9	Program Management	4					
0.1.4	Mission Operation	3					
0.1.5	Payload/Spacecraft	3					
0.1.6	SE&I	3					
0.1.7	Program Management	3					

3.0 COST PLAN FOR RECURRING COSTS

3.1 COST GROUNDRULES

The groundrules used in the development of the recurring cost estimates presented in paragraph 3.4 are as follows:

- Cost estimates are in 1970 dollars at the total cost to the government level and should be considered as rough preliminary estimates.
- It is assumed that a sufficient number of units will be produced to achieve full economics of scale in procurement of components and to pass the point where learning curve effects are applicable to the prime contractor's tasks. It is estimated that this would require production of at least 20 and possibly 50 units.
- Only estimated recurring costs per unit are presented. Furthermore, these costs are applicable to only those units produced after the point at which learning curve effects have become negligible.
- It is assumed that units will be produced on a production line basis with no product improvement or mission oriented changes.
- Costing has been limited to the astrionic module (level 4). Some items of the work breakdown structure which tier up to the astrionic module have been omitted in costing commensurate with the preliminary nature of the study. These items are shown on the cost estimate data forms in paragraph 3.4 with a notation that they have not been costed. Although it is expected that the recovering unit costs for these items would be small as a percentage of the recurring unit cost of the astrionic module as a whole, it must be recognized that the costs shown in the higher levels of the WBS do not include the costs for these items.
- In estimating the recurring costs of the various WBS items the approach was to put the maximum possible portion of the costs against the hardware-oriented items and thus to minimize the costs against task-oriented WBS items such as subsystem installation and checkout. This approach is consistent with the objective of having the costs for an item of hardware represent as nearly as possible the actual cost to the government of that hardware with its related engineering support, quality support, procurement effort, etc.

3.2 CONFIGURATIONS COSTED

Recurring unit cost estimates have been developed for three space tug astrionic module configurations. These configurations are as follows:

- Astrionic Module Configuration for First Tug of Two-Tug Synchronous Orbit Mission
- Astrionic Module Configuration for Second Tug of Two-Tug Synchronous Orbit Mission
- Astrionic Module Configuration for Lunar Landing Mission

These three configurations were chosen to provide a typical range of recurring costs for the ten potential astrionic module configurations studied.

3.3 COST PLAN SUMMARY

The approach used in developing estimated recurring costs for the defined portions of the astrionic module is outlined below:

- A list of major components required to make up each of the three configurations to be costed was generated.

- Estimates of the costs of the major components were developed based on the assumption that all components would be purchased from subcontractors. These estimates were based on costs of similar components purchased in the past, estimates from vendors or engineering estimates.

- Components were grouped into subsystems based on the WBS, and estimated component purchase costs per subsystem were calculated taking into account the number of each component required to make up a complete subsystem.

- The total recurring astrionic module labor costs were developed by multiplying labor costs developed for previous IBM low cost IU proposals by factors to compensate for the increased difficulty of building each of the three astrionic module configurations.

- The total astrionic module recurring labor costs were allocated to the various subsystems in proportion to the percentage of the total component purchase costs attributable to each subsystem.

It is important to keep in mind that the costs presented in paragraph 3.4 are preliminary estimates and that not all items of the astrionic module have been costed. The intent of the cost presentation is to provide available cost information for portions of the astrionic module in a format that will allow other costs developed in the future to be added to these costs in a consistent and orderly manner.

3.4 RECURRING COST ESTIMATES

The "Cost Estimate Data Form" taken from the documents referenced under paragraph 2.1 has been used as the format for the presentation of recurring cost estimates. Commensurate with the preliminary nature of this study, several of the columns on this form dealing with time-phasing of costs, number of units to be produced, etc. have been left blank. Also, as noted in the cost groundrules, the "Expected Cost" column gives the estimated recurring unit cost of the lowest cost units to be produced rather than the expected cost of the first unit to be produced.

Table 3-1 gives the estimated recurring unit costs for items of the astrionic module configuration for the first tug of the two-tug synchronous orbit mission.

Table 3-2 gives the estimated recurring unit costs for items of the astrionic module configuration for the second tug of the two-tug synchronous orbit mission.

Table 3-1. Astrionic Module Estimated Recurring Unit Costs (Synchronous Mission – Reusable 1st Tug)

Identification Number a	WBS Identification b	WBS Level c	No. Units d	Expect. Cost** e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d i	T_s j	Spred Funct k	Learn Index l
0.1.3.2	Astrionics Module	4		7,870*			1				
0.1.3.2.1	Structure	5		240			1				
0.1.3.2.2		5		–			–				
0.1.3.2.3	Astrionics	5		7,370*			2				
0.1.3.2.3.1	Navigation, Guidance & Control	6		3,320			1				
0.1.3.2.3.2	Instrumentation	6		Not Costed			–				
0.1.3.2.3.3	Data Management	6		760			2				
0.1.3.2.3.4	Command & Control	6		460			1				
0.1.3.2.3.5	On-Board Checkout	6		Not Costed			–				
0.1.3.2.3.6	Electrical Power	6		1,830			2				
0.1.3.2.3.7	Electrical Networks	6		590			2				
0.1.3.2.3.8	Thermal Conditioning	6		410			1				
0.1.3.2.3.9	Propellant Utilization	6		Not Costed			–				
0.1.3.2.4	Safety System	5		Not Costed							
0.1.3.2.5	Tooling & STE	5		Not Costed							
0.1.3.2.6	Ordnance System	5		Not Costed							
0.1.3.2.7	Subsystem Installation & Checkout	5		260			1				

**Cost is in $ x 1,000 and is for lowest cost unit produced rather than first unit produced.
* Costs for level 5 & 6 WBS items not costed are missing from these level 4 & 5 totals.

Table 3-2. Astrionic Module Estimated Recurring Unit Costs (Synchronous Mission – Reusable 2nd Tug)

Identification Number a	WBS Identification b	WBS Level c	No Units d	Expect Cost** e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d i	T_s j	Spred Funct k	Learn Index l
0.1.3.2	Astrionics Module	4		8,220*			1				
0.1.3.2.1	Structure	5		240			1				
0.1.3.2.2		5									
0.1.3.2.3	Astrionics	5		7,710*			2				
0.1.3.2.3.1	Navigation, Guidance & Control	6		3,670			1				
0.1.3.2.3.2	Instrumentation	6		Not Costed							
0.1.3.2.3.3	Data Management	6		760			2				
0.1.3.2.3.4	Command & Control	6		460			1				
0.1.3.2.3.5	On-Board Checkout	6		Not Costed							
0.1.3.2.3.6	Electrical Power	6		1,830			2				
0.1.3.2.3.7	Electrical Networks	6		580			2				
0.1.3.2.3.8	Thermal Conditioning	6		410			1				
0.1.3.2.3.9	Propellant Utilization	6		Not Costed							
0.1.3.2.4	Safety System	5		Not Costed							
0.1.3.2.5	Tooling & STE	5		Not Costed							
0.1.3.2.6	Ordnance System	5		Not Costed							
0.1.3.2.7	Subsystem Installation & Checkout	5		270			1				

**Cost is in $ x 1,000 and is for lowest cost unit produced rather than first unit produced
* Costs for level 5 & 6 WBS items not costed are missing from these level 4 & 5 totals

Table 3-3 gives the estimated recurring unit costs for items of the astrionic module configuration for the lunar landing mission.

Table 3-4 is a copy of the definitions of the confidence rating numbers which appear in column "h" on the three cost estimate data forms.

3.5 SUPPORTING ANALYSIS

Table 3-5 provides level 7 data, summed to level 6, which is useful in giving utility to the tiering of flight hardware unit costs and the associated programmatic costs. The costs are summarized to levels 5 and 4 in Table 3-6. The total recurring costs are given in Table 3-7. The $8 million unit cost is an average figure between the costs of astrionic modules for the reusable first and second tugs for the synchronous mission.

4.0 NON-RECURRING COST ESTIMATES

This section presents rough order of magnitude estimates of total space tug astrionic module non-recurring costs. The ground rules for these estimates are delineated in paragraph 4.1, and a summary of the cost figures appears in paragraph 4.2. See Figure 4-1 for the assumed implementation schedule.

Table 3-3. Astrionic Module Estimated Recurring Unit Costs (Lunar Landing Mission)

Identification Number a	WBS Identification b	WBS Level c	No Units d	Expect Cost** e	Highest Cost f	Lowest Cost g	Confid Rating h	T_d i	T_s j	Spred Funct k	Learn Index l
0.1.3.2	Astrionic Module	4		10,800*			1				
0.1.3.2.1	Structure	5		240			1				
0.1.3.2.2		5		–			–				
0.1.3.2.3	Astrionics	5		10,250*			2				
0.1.3.2.3.1	Navigation, Guidance & Control	6		5,330			1				
0.1.3.2.3.2	Instrumentation	6		Not Costed			–				
0.1.3.2.3.3	Data Management	6		1,060			2				
0.1.3.2.3.4	Command & Control	6		460			1				
0.1.3.2.3.5	On-Board Checkout	6		Not Costed			–				
0.1.3.2.3.6	Electrical Power	6		1,850			2				
0.1.3.2.3.7	Electrical Networks	6		1,020			2				
0.1.3.2.3.8	Thermal Conditioning	6		510			1				
0.1.3.2.3.9	Propellant Utilization	6		Not Costed			–				
0.1.3.2.4	Safety System	5		Not Costed			–				
0.1.3.2.5	Tooling & STE	5		Not Costed			–				
0.1.3.2.6	Ordnance System	5		Not Costed			–				
0.1.3.2.7	Subsystem Installation & Checkout	5		310			1				

** Cost is in $ x 1,000 and is for lowest cost unit produced rather than first unit produced.
* Costs for level 5 & 6 WBS items not costed are missing from these level 4 & 5 totals.

Table 3-4. Confidence Level Groups for Cost Estimates

Conf. Level	Estimating Conditions	Nature of the Item	Item Description	Cost Methods and Data
1 Low	Estimating Time and Information Access Completely inadequate amount of time provided to make the estimate or there is a complete lack of access to useful data sources Ground Rules and Assumptions No guidance was provided on ground rules and all assumptions made by the estimator were arbitrary.	State-of-the-Art The item is substantially beyond the current state-of-the-art. Major development work is required. Production Experience No production of any kind has been started.	Specification Status No work on a specification has started. Operating Program Characteristics None of the OPC for using the item have been formulated.	Methods The estimate is almost a poor guess and little or no confidence can be placed in it. Data An almost total lack of current and reliable relevant data makes the cost estimate completely uncertain.
2 Med Low	Estimating Time and Information Access A very short due date or major problems of access to available data tend to make this estimate highly uncertain. Ground Rules and Assumptions Very little guidance was provided relative to ground rules. Most of the assumptions made by the estimator were considered quite arbitrary.	State-of-the-Art The item is slightly beyond the state-of-the-art and some development work will be required. Production Experience Experimental laboratory fabrication of a similar item is in process.	Specification Status Work on a specification is in an early stage and only general requirements are identified. Operating Program Characteristics The general outline of the OPC under which the item will be used has been only tentatively defined and many specific details are lacking.	Methods A highly arbitrary rule-of-thumb has been used. Data The data used to make the estimate is highly suspect, very sparce in quantity, and characterized by major inconsistencies.
3 Med High	Estimating Time and Information Access A more accurate estimate could have been made if freer access or more time had been available to research known data sources. Ground Rules and Assumptions Ground rules were generally adequate. Many of the assumptions were authenticated but a substantial number are considered questionable.	State-of-the-Art The item is within the state-of-the-art but no commercial counterpart exists. Production Experience A prototype of the item has been produced.	Specification Status A specification for the item has not been completed but a specification on a similar item is available. Operating Program Characteristics The general outline of the OPC has been formulated, but many specific details are lacking.	Methods A commonly used rule-of-thumb cost factor, but with no supporting back-up, has been used. Data The data used have been obtained from official or standard sources. Notable inconsistencies, lack or currency, or gaps in data reduce the confidence in the estimate.
4 High	Estimating Time and Information Access There were minor problems of access to available data and there was generally sufficient time to define and cost the item. Ground Rules and Assumptions Major ground rules were provided and most of the assumptions were authenticated.	State-of-the-Art The item will involve a minor modification of commercial or standard aerospace issue items. Production Experience The item has been produced in limited quantity.	Specification Status A specification for the item has been prepared but is under review or revision. Operating Program Characteristics The OPC have been substantially defined, but are under review or revision.	Methods The basic method used to derive the cost is well documented, but no double-check or authentication has been possible. Data The data used are generally relevant and from a reputable source. They are incomplete, preliminary, or not completely current, however.

*Air Force Systems Command Manual, AFSCM 173-1, "Cost Estimating Procedures"

Table 3-5. Supporting Analysis Recurring Hardware Cost – Level 6

WBS Identification	Unit Cost	No. Units	Total Cost
IMU	270	1	270
Laser Radar	945	1	945
Star Tracker	338	2	676
Landmark Tracker	337	1	337
Nav. Guid. & Control Subsystem		Total (Level 6)	2,228
CPU	81	2	162
Bus. Control Unit	27	2	54
Main Memory	81	2	162
Mag. Tape	68	1	68
C. Assn. Unit	34	2	68
Data Mgt. Subsystem		Total (Level 6)	514
USB	162	1	162
VHF	81	1	81
TV Camera	68	1	68
Cmd. & Control Subsystem		Total (Level 6)	311
Fuel Cell	203	2	406
H_2 Tank	202	2	404
O_2 Tank	203	2	406
D.C. Regulator	4	2	8
Electrical Power Subsystem		Total (Level 6)	1,224
SIU	2.7	57	154
Power Dist.	9	1	9
Aux Power Dist.	3	4	12
Wires & Cables	49	1	49
Junction Boxes	1	8	8
Monitoring Unit	0.8	24	163
Electrical Networks Subsystem		Total (Level 6)	395
Coolant Pump	34	2	68
Serv. Heat Exch.	7	1	7
Coolant Accum.	7	1	7
Radiator	7	8	56
Louvers	7	8	56
Mtg. Panel	3	8	24
Misc. Plumbing	20	1	20
Multifoil Insulation	27	1	27
Thermal Conditioning Subsystem		Total (Level 6)	245

Note: $ in Thousands

Table 3-6. Supporting Analysis Recurring Hardware Cost – Levels 4 and 5

WBS Identification	Total Cost
Astrionics	Total (Level 5) 4,917
Structure	Total (Level 5) 135
WBS Hardware Items Not Costed	Total (Level 5) 168
Astrionic Module	Total (Level 4) 5,220
Note: $ in Thousands	

Table 3-7. Supporting Analysis Total Recurring Cost

Total Hardware Costs	$5,220
Total Programmatic Costs	2,780
Grand Total Recurring	$8,000

Notes:
(1) All data is based on an astrionic module configured for the two tug synchronous orbit mission.
(2) $ in thousands.

4.1 COST GROUNDRULES

- Cost estimates are in 1970 dollars at the total cost to the government level.
- Estimates are based on a synchronous orbit mission astrionic module.
- Costs for major component development are not included. It is assumed that all basic components will be developed under other programs and will require only integration and engineering/test for space tug astrionic module application.
- Costs for development of major components which are not available from vendors would have to be added to the figures presented in paragraph 4.2.
- Recurring costs ignore learning curve effects and assume an optimum production rate of 4 to 6 units per year with a large unit buy.
- Costs for post-delivery efforts such as spares, launch support, post-flight evaluation, etc., are not included.

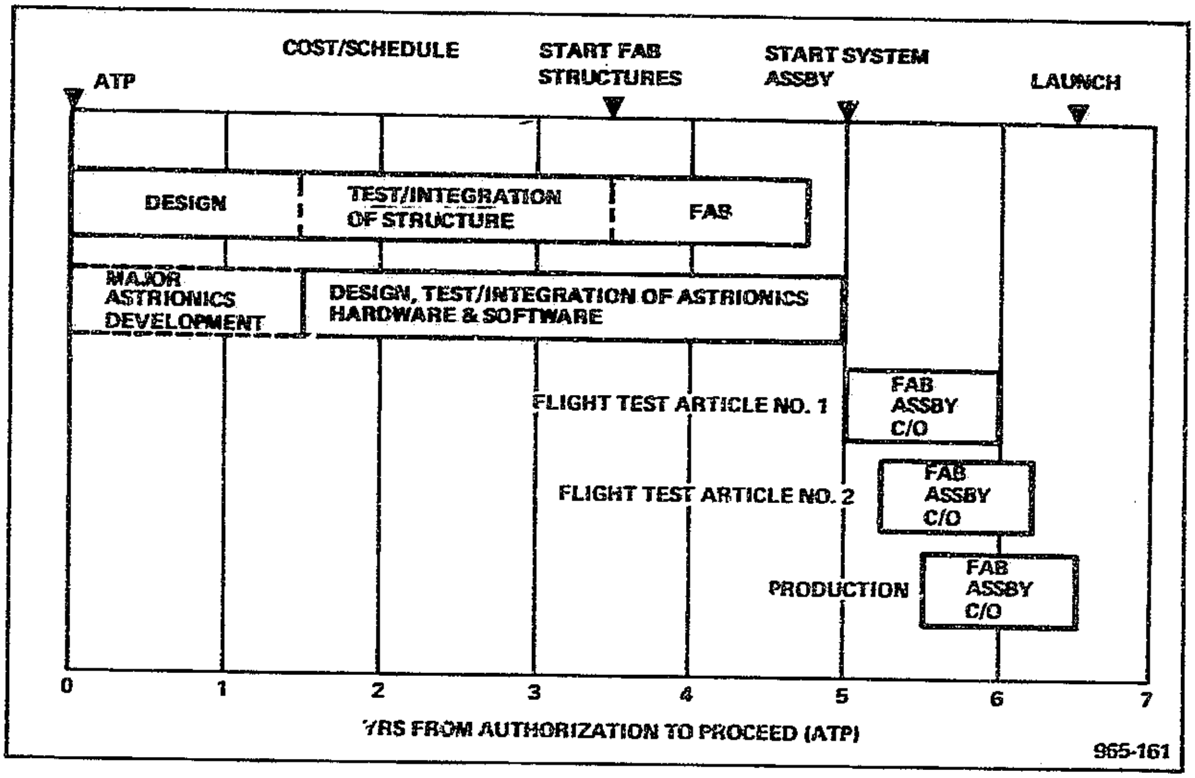


Figure 4-1. Implementation Schedule

- Recurring flight program software costs are assumed to be negligible.
- No costs are included for engineering changes after completion of the first flight test article.
- Authorization to Proceed (ATP) on a 4-1/2 year structure design/test/integration program is assumed to occur 6-1/2 years prior to launch of the first flight test article.
- ATP on a 3-1/2 year astrionics integration/test program is assumed to occur 5 years prior to the first launch.
- Fabrication of the first flight test article is assumed to begin 5 years from original structure ATP with a 12-month build period culminating in delivery of the article 6 months prior to launch.
- Delivery of the second flight test article and subsequent production articles is to follow at intervals of 3 months.
- Non-recurring cost estimates include the costs of 1 static test structure, 1 dynamic test structure, and 1 combination integration and mockup test structure, as well as estimated subcontract static and dynamic testing costs.

- Development cost estimates also allow for five equivalent astrionic systems for non-flight engineering. This includes 2 systems for vibration, altitude and temperature testing; 1 system for qualification; 1 system for engineering model simulation; and 1 equivalent system for engineering laboratory use.

- Two flight test articles are assumed.

- Mission recurring costs are based on ten missions per unit with a ten percent refurbishment factor per mission.

4.2 COST ESTIMATES

Table 4-1 shows the summary program costs. The estimating techniques follow. Note that mission recurring costs assume a 10 mission lifetime and ten percent refurbishment/reconfiguration costs between missions.

Estimating techniques used for development of these rough order of magnitude non-recurring costs are as follows:

- A level of manpower was assumed for the 3-1/2 year astrionics test/integration effort.

- Structure design/test/integration costs are based on INT-21 Vehicle Instrument Unit Structure Study results.

- Hardware costs for astrionics test/integration are based on the application of a factor to the recurring costs presented in Section 3.0 of this appendix.

- Flight test article cost is factored up from estimated recurring unit cost data.

Table 4-1. Summary Program Costs

NON-RECURRING COSTS	FLT TYPE HARDWARE	PROGRAMATIC COSTS	TOTALS
STRUCTURE DESIGN & TEST	$ 1M	$ 6M	$ 7M
ASTRIONIC SYSTEM DESIGN INTEGRATION & TEST	24M	29M	53M
FLIGHT TEST ARTICLES (2)			27M
TOTAL NON-RECURRING			$87M
UNIT RECURRING COST			$ .9M
MISSION RECURRING COST			$1.52M